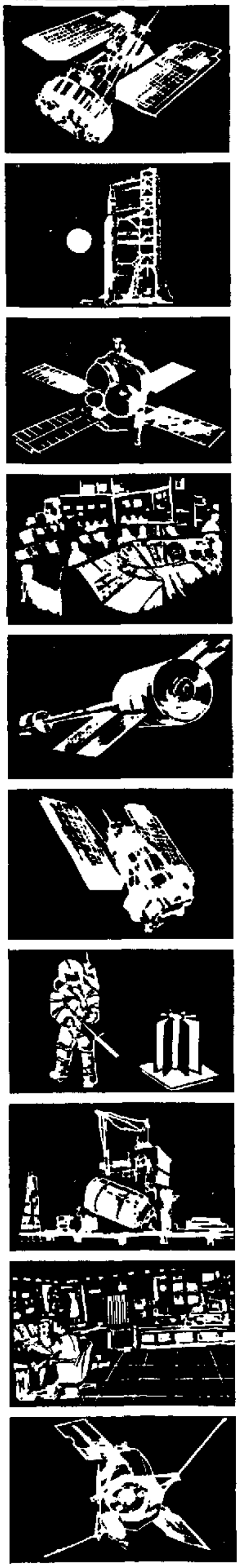
SPACE DIVISION



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16 September 1974

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

Report No. 5
SYSTEM DESIGN AND SPECIFICATIONS

Volume 3
GENERAL PURPOSE SPACECRAFT SEGMENT AND MODULE SPECIFICATIONS

EARTH OBSERVATORY SATELLITE

Prepared for:
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Under
Contract No. NAS 5-20518

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GENERAL ELECTRIC

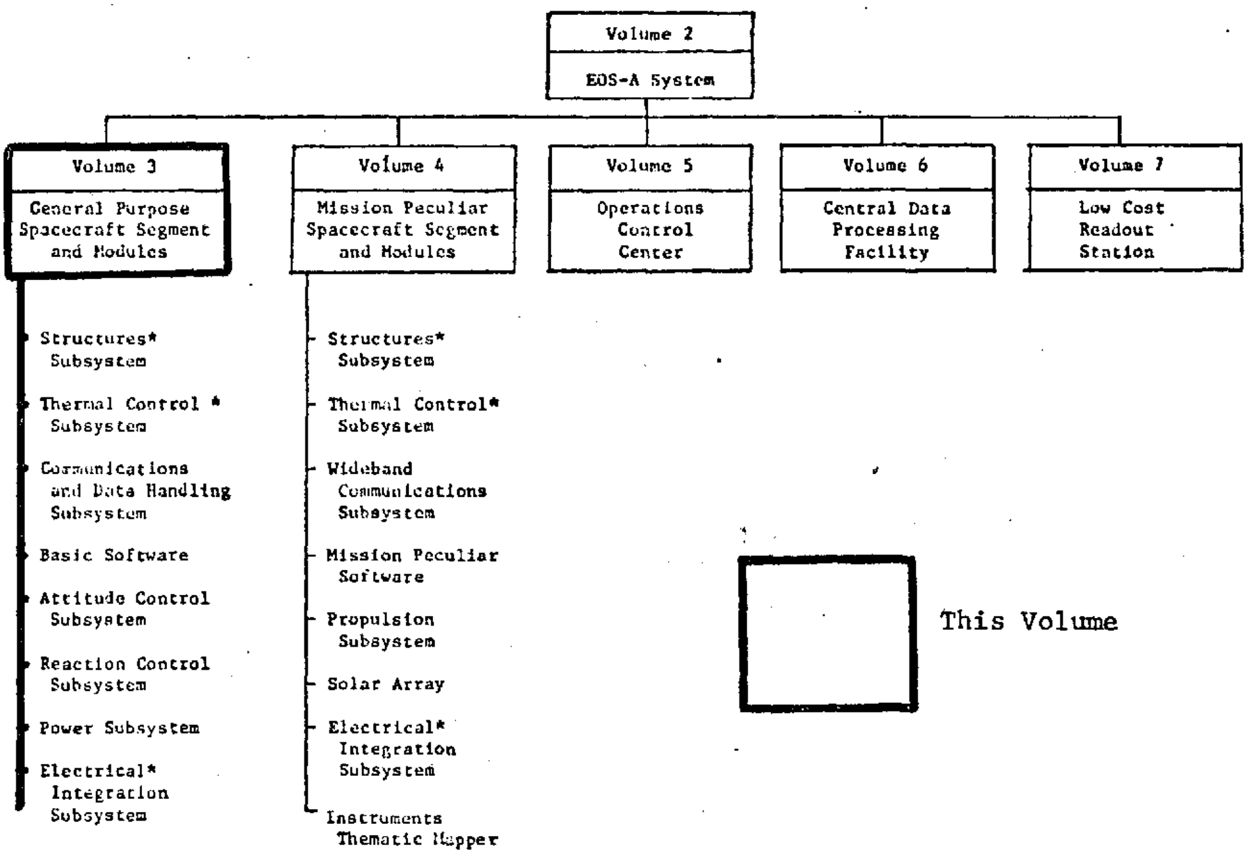
PREFACE

This report, "Baseline System Design & Specifications", has been prepared for NASA/GSFC under contract NAS 5-20518 EOS System Definition Study. It describes the system design that has evolved through a series of design/cost tradeoffs to satisfy a spectrum of mission/system requirements. The basic spacecraft design is compatible with many missions. The EOS-A mission, the potential first mission, is used to define the mission peculiar elements of the system.

For convenience this report is bound in separate volumes as follows:

Volume 1	Baseline System Description
Volume 2	EOS-A System Specification
Volume 3	General Purpose Spacecraft Segment and Module Specifications
Volume 4	Mission Peculiar Spacecraft Segment Specification
Volume 5	Operations Control Center Specification
Volume 6	Central Data Processing Facility Specification
Volume 7	Low Cost Ground Station Specification

Volume 1 "Baseline System Description" presents the overall EOS-A system design, a description of each subsystem for the spacecraft, and the major ground system elements. Volumes 2 through 7 present the specifications for the various elements of the EOS system and are organized according to the specification tree as follows:



* These specifications are written as integral specifications for the GPSS and MPSS and appear in Volume 3 only.

REPORT NO. 5 BASELINE SYSTEM DESIGN & SPECIFICATIONS

VOLUME 3 GENERAL PURPOSE SPACECRAFT
SEGMENT AND MODULE SPECIFICATIONS

SECTION 1
INTRODUCTION

This volume presents the Specifications for the EOS General Purpose Spacecraft Segment and its basic subsystems and modules.

The subsystem and module specifications are presented as "stand-alone" items in order to facilitate their use in further NASA considerations of hardware implementation phases of the EOS program. The specifications contained in this volume and the sections in which they appear are:

SECTION	TITLE
2.0	SPECIFICATION FOR THE EOS GENERAL PURPOSE SPACECRAFT SEGMENT AND MODULES
3.0	SPECIFICATION FOR THE EOS STRUCTURES SUBSYSTEM
4.0	SPECIFICATION FOR THE EOS THERMAL CONTROL SUBSYSTEM
5.0	SPECIFICATION FOR THE EOS COMMUNICATIONS AND DATA HANDLING SUBSYSTEM MODULE
6.0	SPECIFICATION FOR THE EOS BASIC SOFTWARE
7.0	SPECIFICATION FOR THE EOS ATTITUDE CONTROL SUBSYSTEM MODULE
8.0	SPECIFICATION FOR THE EOS HYDRAZINE REACTION CONTROL SUBSYSTEM
9.0	SPECIFICATION FOR THE EOS POWER SUBSYSTEM MODULE
10.0	SPECIFICATION FOR THE EOS ELECTRICAL INTEGRATION SUBSYSTEM

SECTION 2.0

SPECIFICATION NO. SVS-XXXX

16 Sept. 1974

SPECIFICATION
FOR THE
EARTH OBSERVATORY SATELLITE (EOS)
GENERAL PURPOSE SPACECRAFT SEGMENT
AND
MODULES

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1.0 SCOPE

This specification establishes the requirements for performance and design, qualification and acceptance testing of an Earth Observatory Satellite (EOS) General Purpose Spacecraft (GPSS).

The GPSS is configured to provide a modular space platform which has the capability and flexibility to support any number of programs requiring the use of a 3-axis stabilized earth orbiting satellite. The basic concept of the GPSS is to use this "standard" spacecraft plus unique missions adaptions and equipment for a variety of space missions.

This specification describes the subsystems and modules which make up the total GPSS. These items include the structural, thermal control, communications and data handling, basic software, attitude control, reaction control, power and electrical subsystems.

2.0 APPLICABLE DOCUMENTS

The following documents of the exact issue shown, form a part of this specification to the extent specified herein. In the event of conflict between documents referenced here and the detail requirements in the following sections, the detail requirements of this specification shall supersede. In the event of conflict between documents referenced here and lower tier references to documents referenced here, the former supersede.

2.1 Applicable Documents

SPECIFICATIONS

National Aeronautics and Space Administration

EOS-410-02	Specifications for EOS System Definition Studies, 13 September 1974
S-311-P-11	Quality Monitoring of Integrated Circuits , 1 June 1970
S-323-P-10	Connectors, Subminature Electrical and Coaxial Contacts for Space Flight Use, Revised December 1969

MILITARY

MIL-C-38999	Connectors, Electrical, Miniature, Quick Disconnect, Est. Reliability
MIL-C-29012A	Connectors, Coaxial, RF, General Specification for
MIL-C-26482	Connectors, Electric, Circular, Miniature, Quick Disconnect
MIL-C-17	Cables, RF, Coaxial, Dual Coaxial, Twin Conductors, Twin Head
MIL-W-18044	Wire, Electric Cross-linked, Polyalkene, Insulated, Copper
MIL-E-5400K	Electronic Equipment, Airborne, General Specification for

General Electric

SVS XXXX	Specification for EOS Communications and Data Handling Subsystem Modul_
SVS XXXX	Specification for EOS Power Subsystem Module and Solar Array
SVS XXXX	Specification for EOS Attitude Control Subsystem Module
SVS XXXX	Specification for EOS Reaction Control Subsystem
SVS XXXX	Specification for EOS Structure Subsystem
SVS XXXX	Specification for EOS Thermal Subsystem
SVS XXXX	Specification for EOS Basic Software
SVS XXXX	Specification for EOS Electrical Integration Subsystem

Standards

National Aeronautics and Space Administration

Aerospace Data Systems Standard, Part II, Section 3, PCM Command Data System Standard, prepared by GSFC Data Systems Requirements Committe, 23 April 1968.

Aerospace Data System Standards, Part III, Associated System Standards, Section 2, "Spacecraft Clock Systems Standard", prepared by GSFC Data Systems Requirements Committee, April 1964.

Aerospace Data System Standards, Part III, Associated Standards, Section I "Radio Frequency and Modulation Standard for Space-to-Ground Telemetry", prepared by GSFC Data Systems Requirements Committee, November 1965.

Aerospace Data Systems Standard, Part I, Telemetry Standards, Section 1, Pulse Code Modulation Telemetry Standard, prepared by GSFC Data Systems Requirements Committee, January 27, 1966.

Part III, Associated Systems Standards, Section 3 "Spacecraft Minitrack Signal Source Standard", prepared by GSFC Data Systems Requirements Committee, October 1963.

MILITARY

MS33540C	Safety Wiring, General Practices for
MIL-STD-454B	Standard General Requirements for Electronic Equipment
MIL-STD-143A Change 1	Specification and Standards, Order of Precedence for selection of
MS-33586A	Metal, Definition of Dissimilar
MIL-STD-130C	Identification Marking of US Military Property
MIL-STD-1247A	Identification of Pipe, Hose, and Tube Lines for Aircraft, Missile and Space Systems

OTHER PUBLICATIONS

National Aeronautics and Space Adminsistration

NHB 5300.4 (3A) May 1968	Requirements for Soldered Electrical Connections
PPL-12 Latest Issue	GSFC Preferred Parts List
NHB 5300.4 (1A)	Reliability Program Provisions for Space Systems Contractors
NHB 5300.4 (1B)	Quality Assurance Program Provisions for Space Systems Contractors.

Air Force Manuals

AFM 71-4	Air Force Regulations for Transportation of Exposive and Other Dangerous Material
AFWTRM127-1	Air Force Western Test Range Safety Manual

MILITARY HANDBOOKS

MIL-HDBK-5A	Metallic Materials and Elements for Aerospace Vehicle Structure
MIL-HDBK-17	Plastics for Flight Vehicles

INTERSTATE COMMERCE COMMISSION

T.C. George's Tariff No. 6C	Transportation of Explosives and Other Dangerous Articles by Commerical Aircraft
T.C. George's Tariff No. 19	Transportation of Explosives and Other Dangerous Articles by Land, Water, in Rail Freight Service and by Motor Vehicle (Highway) and water.

GENERAL ELECTRIC COMPANY

XXXXX	EOS General Purpose Spacecraft Quality Program Plan
XXXX	Configuration Management Plan for EOS General Purpose Spacecraft
XXXX	Reliability Program Plan, EOS General Purpose

3.0 REQUIREMENTS

3.1 General

3.1.1 Mission Objectives

The EOS General Purpose Spacecraft is designed to provide a modular space platform which has the capability and flexibility to support any number of programs requiring the use of 3-axis stabilized earth orbiting satellite.

The General Purpose Spacecraft provides attitude stabilization, electrical power and a communications and data handling subsystem which can support various mission peculiar subsystems. The basic concept of the General Purpose Spacecraft is to provid a cost-effective means for implementation of many future space programs including the use of the Space Shuttle recovery capabilities.

3.1.2 Basic Requirements

The EOS General Purpose Spacecraft is designed to meet the following basic requirements.

a) Provide a basic structural/mechanical subsystem to support the required modules throughout the launch and orbit environment.

b) Provide a thermal control system which is self sufficient and will not require or cause interaction with the mission peculiar section of the satellite.

c) Provide a 3-axis stabilized Attitude Control System Module with a pointing accuracy of 0.007 degrees, a rate of 5×10^{-5} deg/sec and a position accuracy knowledge of 0.007 degrees.

d) Provide a Power Subsystem Module and solar array which is capable of supplying all power required for the operation of the basic General Purpose Spacecraft as well as 200 watts orbit average to the mission peculiar payload.

e) Provide a Communications and Data Handling Subsystem Module which provides the capability for tracking the spacecraft, ground and on-board control of all spacecraft and payload functions, and retrieval of spacecraft housekeeping, narrow-band, and medium-band data.

3.1.3 Orbit Design Restraints

The EOS General Purpose Spacecraft must be capable of performing in earth orbits ranging from a minimum of 250 n.m. to 19,323 n.m.

3.1.4 System Relationship

The General Purpose Spacecraft is to be part of a total earth orbiting satellite system and is considered to be the means by which the mission peculiar segment of the satellite is supported and powered during the pre-launch, launch and orbit environments. Along with an Operations Control Center, a Data Processing Facility and attendant Launch Vehicle Operations, the General Purpose Spacecraft makes up a total program system which is capable of being varied considerably from mission to mission. The modular/flexible design of the General Purpose Spacecraft allows it to perform this function in a large variety of cases.

3.2 SPACECRAFT SYSTEM PERFORMANCE

3.2.1 Spacecraft Service Subsystems

3.2.1.1 Structural Subsystem

The EOS General Purpose Spacecraft modular design is the "aft" subsystem or "Bus" section of a total spacecraft assembly. The General Purpose segment consists of a core structure supporting the Attitude Control System Module, the Power Module and Solar Array, the Solar Array Drive, the Communications and Data Handling Subsystem Module, and the Propulsion Module as required. The structure provides the capability to support a maximum of 2400 pounds of payload equipment in addition to the spacecraft service subsystems.

3.2.1.2 Thermal Subsystem

The thermal environment is passively controlled to provide a temperature range of $70 \pm 5^{\circ}F$ for the ACS and C&DH modules and $50 \pm 5^{\circ}$ F for the power module. Each module is thermally independent and is designed not to interact with the thermal response of the other modules or subsystem structure.

3.2.1.3 Communications and Data Handling Subsystem Module

Table 3.2.1.3-1 lists the assigned frequencies, bandwidth and modulation for the assigned data links.

Table 3.2.1.3-1. C&DH Frequency Allocations

Data Link	Use	Frequency (MHz)	Modulation
USB Uplink	CMD, GRARR	2050-2150	[PCM/PSK + GRARR] / PM
USB Downlink	NBT, GRARR & Medium Rate TLM	2200-2300	Composite/PM
EOS to TDRSS	Ranging, NBT & Medium Rate TLM	2200-2300	Composite/PM
TDRSS to EOS	Ranging & Command	2025-2120	-

3.2.1.3.1 Command

Command capability is provided which is compatible with the Standard Stadan System, conforming with Aerospace Data Systems Standard, Part II, Section 3, PCM Command Data System Standard, prepared by GSFC Data Systems Requirements Committeė, 23 April 1968. Digital commands shall be provided which can be executed in real time or from storage. Provisions for override of stored commands and for verifying the stored command load shall be provided.

Any occurance of erroneous commands, false commands or interruptions during command loading shall not result in a mission termination. The system design shall preclude any catastrophic result (inability to retrieve) because of failure to execute or in-advertent execution of any single command.

3.2.1.3.2 Clock

Timing signals and a spacecraft clock are generated for distribution to payload and spacecraft service subsystems. The time code is generated relative to Universal Time and conforms to the Aerospace Data Systems Standards, Part III, Associated Systems Standards, Section 2, "Spacecraft Clock Systems Standard", prepared by GSFC Data Systems Requirements Committee, April 1964. The clock update time shall be known to within 1-bit timing of the PCM telemetrydata rate.

3.2.1.3.3 Narrow Band Telemetry

Capability is provided to transmit spacecraft service subsystem and payload performance data through either the TDRSS or a USB transponder. The telemetry link conforms to the Aerospace Data Standards, Part III, Associated Standards, Section I, Radio Frequency and Modulation Standard for Space to Ground Telemetry, prepared by GSFC Data Systems Requirements Committee, November 1965. Analog data is quantized to 8-bits per sample and conforms to Aerospace Data Systems Standard, Part I, Telemetry Standards, Section I, Pulse Code Modulation Telemetry Standard, prepared by GSFC Data Systems Requirements Committee, January 1966.

Realtime data transmission of data at 32K bits per second is via the TDRSS, the USB transponder or both simultaneously. Playback transmission of stored data is 640K bits per second is via the USB transponder or TDRSS. Backup capability is provided to transmit the playback data via the TDRSS. Provision is made for realtime transmission simultaneously with playback transmission via the USB transponder and TDRSS.

The bit error rate for primary mode telemetry links shall not exceed 1 in 10^6 with typical ground station parameters assumed as delineated in paragraph 3.3.4.3a, and 6db **system margin.**

3.2.1.3.4 Wideband Telemetry

The spacecraft transmits TM and MSS data in real time thru TDRSS and also thru two identical STDN S/C to ground links simultaneously. A separate link is supplied for Local User data and transmits either MSS or compacted TM data. The RF bandwidth allotment is 227 MHz for TDRSS and 375 MHz for STDN and Local User's combined.

3.2.1.3.5 Tracking

The spacecraft provides a coherent USB transponder for GRARR compatibility. A Ku band transponder is also provided for tracking via TDRSS.

3.2.1.4 Attitude Control Subsystem Module

With maxiumum initial rates of one (1) degree/sec. about each axis, and from any initial attitude, the spacecraft shall acquire the sun within $\pm$ 1 degree with rates less than 0.5 degree/hr. The S/C will then determine its attitude to within 1 degree in 20 minutes and to within 36 arc seconds in 3.3 hours. The spacecraft shall be stabilized to within the accuracy specified in paragraph 3.2.1.4.2 within 3 orbits after star acquisition.

3.2.1.4.2 Operational Control

The spacecraft shall be stabilized to within $\pm$ 36 arc seconds (1σ) yaw axis to the local vertical (roll and pitch axes), and within $\pm$ 36 arc second (1σ) roll axis to the orbit plane (yaw axis); during steady state condition. Steady state rates shall not exceed 10^{-6} degrees/second (each axis) for a period of 30 minutes. The S/C shall not exhibit a jitter of more than .0006 degrees for a period less than 20 minutes nor .0003 degrees for periods of less than 30 seconds.

3.2.1.4.3 Attitude Sensing

The spacecraft shall provide via narrow band telemetry the pitch and roll attitude of the spacecraft to an accuracy of $\pm$ 36 arc seconds, and the yaw attitude to an accuracy of 36 arc seconds.

3.2.1.5 Power Subsystem Module and Solar Array

The General Purpose Spacecraft Power Subsystem Module shall be designed to permit normal operation of the S/C subsystems as well as payload operation. The power subsystem capability shall not restrict payload operation, either real time or store and playback, under the following constraints:

a. Data Acquisition Stations specified in paragraph TBD with 5^{o} elevation or local terrain masking, whichever is greater.

b. Earth orbits ranging from 250 n.m. to 19323 n.m.

c. All payload and service subsystems operating normally at the end of one year.

3.2.1.6 Propulsion Module

The Propulsion Module has the combined capability of performing the spacecraft functions of reaction control, orbit adjust, and orbit transfer. This specification addresses only the reaction control section of the Propulsion Module.

The Reaction Control System must provide the following functions:

Initial Stabilization & Restabilization	-	400 lbs-sec.
Backup Momentum Unloading	-	2275 lbs-sec.

3.2.2 AGE Subsystems

3.2.2.1 Transportation

The Spacecraft Transportation Equipment provides for protection of the General Purpose Spacecraft during transportation over the road or by air. Within its container, the spacecraft will not be subjected to environments in excess of those specified in paragraph 3.4.4.

3.2.2.2 Fuel Servicing

The fuel service equipment will provide for loading 18.8 lbs. of Hydrazine and .6 lbs. of GN_2 at 660 psi for the reaction control system.

3.2.2.3 Mass Property

Mass Property measurement equipment will provide capability for (a) locating the c.g. within $\pm$ 0.020 inches (x and y axes) and $\pm$ 0.050 inches z (axis), (b) determining spacecraft weight within $\pm$ 0.1% and (c) determining the products of inertia to within the following:

I_{xz}	(roll-yaw)	0.85 slug-ft^2
I_{yz}	(pitch-yaw)	3.0 slug-ft^2
I_{xy}	(roll-pitch)	No limit

3.2.2.4 Alignment

Alignment equipment will provide capability to permit alignment of spacecraft mounted equipment as specified in paragraph 3.5.1.2.

3.2.2.5 Electrical Test

Electrical test equipment will provide capability to control and monitor spacecraft system and subsystem operation during system and subsystem levels of testing described as specified in paragraphs (later) . The equipment shall be designed to protect the spacecraft from damage in the event of a malfunction in the AGE. Control of attitude control stimulators and targets shall be provided.

3.2.2.6 Ground Station

A test ground station will provide capability to generate and verify commands through both STDN and TDRSS links, to receive, process, record and display narrowband telemetry data and tracking data.

3.2.2.7 Environmental Support

Equipment necessary to supply power and control heater arrays and temperature monitoring equipment will be provided for vacuum thermal testing.

3.3 SYSTEM INTERFACE REQUIREMENTS

3.3.1 Launch Vehicle

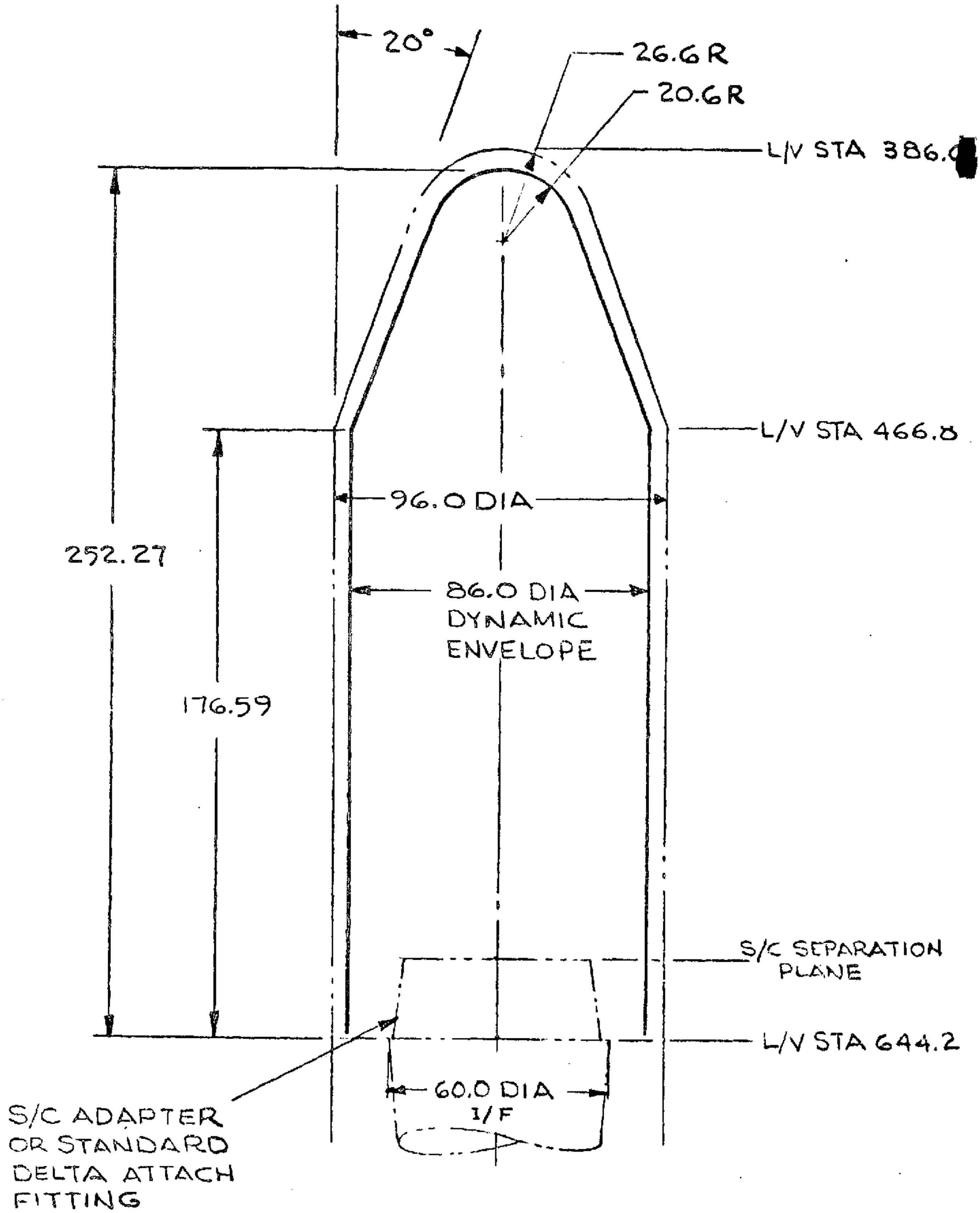
The General Purpose Spacecraft plus a Mission Peculiar Segment will be launched into orbit by the Delta 2910 Launch Vehicle and shall be compatible with the launch vehicle interfaces indicated in the following paragraphs. It will also be capable of being Shuttle retrievable.

3.3.1.1 Flight Loads

The spacecraft shall withstand the launch vehicle induced flight loads as given in Section 3.4.4.

3.3.1.2 Mechanical Interface

The General Purpose Spacecraft plus a Mission Peculiar Segment shall fit within the shroud/spacecraft envelope shown in Figure 3.3.1.2-1 . These clearances shall be maintained during static conditions and while the spacecraft is subjected to the flight loads defined in Section 3.4.4 .



DELTA LAUNCH VEHICLE
FAIRING ENVELOPE

FIGURE 3.3.1.2-1

3.3.1.3 Access

The launch vehicle shroud shall provide access doors as required for several spacecraft ground functions.

3.3.1.4 Electrical Interface

3.3.1.4.1 Ground Monitoring

The launch vehicle/spacecraft interface shall provide wiring and connectors for up to TBD pins for hardwire ground monitoring of spacecraft functions from installation of the spacecraft until liftoff. This wiring shall be routed from the block house to the launch vehicle ground umbilical and up through the spacecraft adapter-launch vehicle transition ring.

3.3.1.4.2 Interface Pin Assignments

The launch vehicle/spacecraft interface shall provide for ground monitoring and launch phase telemetry, electrical power and event functions. Wiring and connectors for these functions shall be fed through the spacecraft adapter-launch vehicle transition ring.

3.3.1.4.3 Electroexplosive Device (EED) Requirements

Launch vehicle shall provide power to fire four electroexplosive bolt cutters to effect spacecraft separation. 150% all-fire current shall be provided. The EED's have the following characteristics:

a. Recommended Fire Current: 5.0 amperes minimum for 20 milliseconds per squib*

b. No-fire Currents: 1.0 amperes maximum for 5 minutes per squib*

c. Bridgewire Resistance: 0.25 ± 0.1 ohms per squib*

d. All-fire Current: 2.58 amperes minimum per squib*

* Per squib means per bridgewire

3.3.1.4.4 RF Transmissibility

Should RF communications links (as shown in Table 3.3.1.4.4) between spacecraft and the NASA telemetry area at the launch site be necessary during the time the spacecraft is installed on the launch vehicle at the launch pad re-radiation antennas will be provided and hardwired to the telemetry area or the shroud will be modified with windows. If the windows are used, the losses through the windows should be less than 3 db for frequencies between 130 and 2300 MHz.

Table 3.3.1.4-4. S/C RF Signature.

Function	Nominal Frequency
TDRSS Transmitter	2200 MHz
TDRSS Command Receiver	2050 MHz
USB Narrowband Transmitter	2200 MHz
USB Command Receiver	2050 MHz

3.3.1.5 Environmental Control

3.3.1.5.1 Thermal Control

The launch vehicle shall be sealed at the spacecraft interface to prevent air flow from the launch vehicle to the spacecraft. Cooling air at an inlet temperature of 50 to 66°F shall be provided within the shroud to control the spacecraft environment on the launch pad to lift-off. Cooling air flow rate shall be at least 50 pounds per minute. The relative humidity of the cooling air shall not exceed 50%. The S/C shall be provided an environment equal to or better than class 100,000 at all times.

3.4 SYSTEM DESIGN REQUIREMENTS

3.4.1 Reliability

The General Purpose Spacecraft is designed to optimize system reliability. Functional and block redundancy are utilized to enhance system reliability in critical areas.

3.4.2 Maintainability

The General Purpose Spacecraft shall be designed for ease of maintainability to minimize downtime during assembly, test, and checkout.

3.4.2.1 Maintenance Requirements

a. The General Purpose Spacecraft shall be designed for removal and replacement down to the module level.

b. All electrical parameter trimming, mechanical adjustment or alignment performed at the subsystem or lower level shall be performed prior to subsystems Flight Acceptance testing.

c. Spacecraft system level electrical trimming, mechanical adjustment or alignment shall be such that these parameters can be re-established in the event of subsequent disassembly and assembly of the spacecraft.

3.4.2.2 Maintenance and Repair Cycles

The General Purpose Spacecraft shall be designed such that no scheduled maintenance will be required after the hardware has been shipped to the launch site with the exception of the normal servicing functions associated with propellant loadings, Electroexplosive Device (EED) installations and battery conditioning. Repairs to be effected at the launch site shall be limited to those failures or malfunctions which are discovered at the launch site and shall only be accomplished by replacement of equipment at the module level.

a. Access shall be made to all module test plugs, harness break-in points, and pressurant and propellant fill and drain valves through openings, access ports or doors without disassembly of the spacecraft.

b. Optical references located on critically aligned components shall be externally visible through inspection ports or access doors.

c. Access shall be provided for normal servicing before installation of thermal blankets such as battery, EED removal and installation, and the capping of critical optical or pneumatic components.

d. Access shall be provided for installation and subsequent removal of all non-flight hardware before launch.

3.4.3 Useful Life

The useful life of the spacecraft shall be a minimum of 1 year starting with the acceptance of the spacecraft by the procuring agency. The useful life of individual subsystems and modules within the spacecraft shall include sufficient additional time to allow for the elapsed time during transportation, handling, storage and testing phases prior to acceptance by the procuring agency. The orbital operational life shall be a minimum of two years.

3.4.4 Environmental

The observatory shall suffer no performance degradation beyond the limits specified elsewhere in this document while exposed or after exposure to the environmental conditions specified in Table 3.4.4-1. This table lists the environmental requirements for each phase of spacecraft life from transportation through orbital operation plus environmental conditions to which the equipment shall be exposed during Qualification and Acceptance Tests. Environmental criteria supplementary to Table 3.4.4-1 is contained in Sections 3.4.4.1 through 3.4.4.5.

PHASE ENVIRONMENT	X-PORT	HANDLING & ASSEMBLY/TEST & CHECKOUT	PRE-LAUNCH	LAUNCH	ORBITAL	ENVIRONMENTAL TESTING QUAL.	ACCEPT.
THERMAL (°F)	+35 to +100	72 ±3 (Controlled) 65 ±15 (Uncontrolled)	50 to 66	(TBD) INSIDE SHROUD SURFACE 200	MSF:440±13 Albedo 167 Earth Emitt.67 (BTU/HR/FT^2)	N/A	TBD
PRESSURE (mm Hg)	300 to 790	760 ± 30	760 ±30	780 to 1X10^{-18}	1X10^{-18}	790 to 1X10^{-5}	790 to 1X10^{-5}
HUMIDITY %RH	55	45 to 55	35 to 55	NA	NA	Not Req'd	Not Req'd
VIBRATION	See Para. 3.4.4.1.2		NA	See Para. 3.4.2.2.2	NA	See Para 3.4.2.2.3	See Para 3.4.4.2.3
SHOCK	See Para. 3.4.4.1.3		NA	See Para. 3.4.4.5	NA	See Para 3.4.4.4	See Para 3.4.4.4
ACCELERATION	See Para. 3.4.4.1.1	See Para. 3.4.4.1.1	NA to Z Axis	See Para. 3.4.4.2	TBD	Not Req'd	Not Req'd
ACOUSTIC NOISE	NA	NA	NA	See Para. 3.4.4.4		See Para 3.4.4.3	See Para 3.4.4.3
PARTICLE BOMBARD	NA	NA	NA	Table 3.3.4.-2	Table 3.4.4-2	Not Req'd	Not Req'd
METEROID	NA	NA	NA	NA	Table 3.4.4-3	Not Req'd	Not Req'd

Table 3.4.4-1. Environmental Criteria

The General Purpose Spacecraft will operate during portions of Assembly, Test and Checkout, Pre-Launch, Launch, and Orbital phases. The spacecraft will not operate during the Transportation phase.

3.4.4.1 Shipping, Handling, and Transportation

3.4.4.1.1 Acceleration

During the shipping and handling, equipment shall be capable of experiencing limit loads of up to ±3 g in any direction.

3.4.4.1.2 Vibration

Equipment in its shipping containers shall be capable of withstanding the following sinusoidal vibration environments:

Frequency (Hz)	Acceleration (g; 0-Peak)	Displacement (Inches, D.A.)
2-5	±0.375	
5-1000	±1.3	0.30

3.4.4.1.3 Shock

Equipment in its shipping containers shall be capable of withstanding a 3 G maximum acceleration in any axis.

3.4.4.2 Structure Performance

3.4.4.2.1 Support

The General purpose spacecraft shall be capable of supporting a maximum gross weight, above the launch vehicle interface plane of 3500 pounds (including itself and a mission perculiar segment) with a weight breakdown as shown in Table 3.4.4.2.1-1.

TABLE 3.4.4.2.1-1

GENERAL PURPOSE SPACECRAFT WEIGHT BREAKDOWN

	S/S	TOTALS
TOTAL BASIC S/C		1044
STRUCT/MOD/MECH	360	
ACS	90	
POWER	222	
C & DH	184	
HARN. & SIGN. COND.	110	
THERMAL	38	
PNEUMATICS	40	

3.4.4.2.2 Stiffness

For powered flight, the primary and secondary structures shall provide adequate stiffness to satisfy the minimum resonant frequency requirements in Table 3.4.4.2.2-1. For this evaluation, the spacecraft and modules shall be analyzed in the launch configuration cantilevered from their attachment points.

In the orbital configuration, a TBD Hertz minimum resonant frequency for appendages shall be adequate to preclude dynamic interaction between the structure and the Attitude Control System. Analytical evaluations of the deployed solar arrays, including in detail the effects of hinge flexibility and solar drive flexibility, shall be used to demonstrate compliance with this requirement.

3.4.4.2.3 Strength

A. For preliminary sizing, the primary structure shall be designed to the qualification level steady-state accelerations of Table 3.4.4.2.3-1. Off-loading conditions for the Delta configuration shall be considered in the analysis. Subsequent dynamic analyses shall determine responses to the qualification vibration test levels of Tables 3.4.4.2.3-2 and 3.4.4.2.3-3. including estimated "notching" levels to prevent excessive dynamic test loads. In these response analyses, a modal damping ratio of C/C_c = 0.05 will be used for the primary structure modes.

Table 3.4.4.2.1-1

Spacecraft Minimum Resonant Frequencies

During Powered Flight

A. Primary Structure

Structure \ Launch Vehicle	THOR/DELTA Lat. (Hz)	THOR/DELTA Long. (Hz)	TITAN III B Lat. (Hz)	TITAN III B Long. (Hz)	SHUTTLE Lat. (Hz)	SHUTTLE Long. (Hz)
Spacecraft *	10	30	10	30	TBD	
Subsystem Module **	60	60	60	60	60	60
Experiment ***	TBD		TBD		TBD	

* Cantilevered from the launch vehicle/adapter interface.

** Mounted on ball joints at the four attachment points.

*** Cantilevered from the transition ring attachment points.

B. Secondary Structure

Item	Longitudinal (Hz)	Lateral (Hz)
Instrument Section Structure	70	70
Antenna Mounting	70	70
Stowed Solar Array Module Mounting	25-30	8
Stowed Solar Array Panel	70	15
Subsystem Component Mounting	100	100

Table 3.4.4.2.3-1

Qualification Level Quasi-Steady Accelerations

Launch Vehicle & Condition	Longitudinal (g)	Lateral (g)
Delta		
Max. Lateral (Lift-off)	- 4.4	± 3.0
Max. Compression (MECO/POSO)	-18.0	± 1.0
Max. Tension	1.5	± 3.0
Titan III B/NUS		
Max. Lateral (Lift-off)	- 2.9	± 2.5
Max. Compression (Stage II Shutdown)	-13.5	± 1.3
Max. Tension (Stage I Shutdown)	3.1	± 1.9
Shuttle		
Lift-off	- 3.5	± 1.3
Orbiter End Burn	- 5.0	± 0.6
Entry	0.4	4.5
Landing & Braking	± 2.3	3.8
Crash (Ultimate Applied Separately)	9.0 - 1.5	4.5 - 2.0

Table 3.4.4.2.3-2. Sinusoidal Vibration

Axis \ Launch Vehicle	Delta		Titan III B		Space Shuttle	
	Frequency (Hz)	Acceleration (G, o-pk)	Frequency (Hz)	Acceleration (G, o-pk)	Frequency (Hz)	Acceleration (G, o-pk)
Longitudinal Axes	5-15* 15-21 21-100	2.3 6.0 2.3	5-20* 20-50 50-200	9.0 in/sec 3.0 2.3		TBD
Lateral Axes	5-14* 14-100	2.0 1.5	5-22* 22-200	2.0 1.5		TBD

Sweep Rate: 2 octaves/minute

* Limited with the performance of the exciter. The amplitude in these frequency ranges shall not exceed 0.5 inches D.A.

Table 3.4.4.2.3-3. Random Vibrations*

	Frequency (Hz)	PSD (G^2/Hz)	GRMS	TIME (Seconds Per Axis)
Thor Delta	20-300 300-700 700-2000	+4 db/Oct .16 -3 db/Oct	14.1	20
	20-300 300-700 700-2000	+4 db/Oct .07 -3 db/Oct	9.5	70
Titan III D	20-250 250-2000	+6 db/Oct .16	17.0	240
Shuttle	20-100	+6 db/Oct .65 -6 db/Oct	24.3	90

* Thrust and Lateral Axes

The final design loads shall be determined in the subsequent phase from coupled launch vehicle-spacecraft dynamic response analysis and dynamic analyses of other critical conditions. These analysis results shall be used to determine the notching factor for the sinusoidal vibration tests at the primary spacecraft resonances such that the primary structure stress levels experienced during flight shall be compatible with those experienced during the vibration test.

B. Steady State Accelerations.

a. Hoist and Spacecraft to L/V mating

Nz	Nx	Ny
-3.0	±0.5	±0.5

b. Transportation (Rail, Air, Motor, and Water)

Vertical: 2.0 g up
4.0 g down

Lateral (sideways): ± 2.5 g

Longitudinal: ±3.0 g (due to docking ramp impact)

These loads are maximum expected equivalent static loads due to carrier operation. They are to be applied separately. The directional terms are with respect to the carrier motion. These loads are to be reacted by the appropriate transportation support configuration. Special procedures, handling equipment, transportation support configuration and shock and vibration isolation between spacecraft and carrier floor will be utilized in order not to exceed the above loads and that the spacecraft element loadings do not exceed 50 % of the flight qualification loads.

The design load factors of safety shown in Table 3.4.4.2.3-4 applied to qualification loads presented in Table 3.4.4.2.3-1 to obtain the structural design yield and design ultimate loads.

The pressure vessel factors shown in Table 3.4.4.2.3-5 shall be applied to maximum expected operating pressures to obtain design pressures for all hydraulic and pneumatic components.

Table 3.4.4.2.3-4 Design Load Factors of Safety

Load Condition	Design Lead Factors of Safety	
	Yield	Ultimate
Launch (Qualification Level)	1.5	2.0
Orbital (Qualification Level)	1.5	2.0
System Qualification Test	1.5	2.0
Transportation, Handling (Apply Load Factors to Loads of Paragraph 3.2.1.2.3.1	1.5	2.0

Table 3.4.4.2.3-5 Pressure Vessel Factors

Pressure Container	Operating	Proof	Burst
Main Propellant Tanks	1.00	1.50	2.00
Vessels Including Accumulators & Pressurization Bottles	1.00	1.50	2.50
Hydraulic & Pneumatic Lines, Fittings & Hoses	1.00	2.50	4.00
Propellant Supply and Vent Components	1.00	1.50	4.00

The General Purpose Spacecraft when subjected to the environments , the design factors of safety presented herein shall maintain the minimum design margins of safety presented in Table 3.4.4.2.3-6

Table 3.4.4.2.3-6 Minimum Margins of Safety

	Minimum Margin of Safety
Fasteners in Shear	+.15
Bolts in Tension	+.50
Fittings	+.15
Lugs	+.25
Welds - Electron Beam	+.15
Welds - Other	+.50 (Dependent on Inspection Procedure)
Bonded Joints	+.50

Margins of safety less than 2.0 shall be indicated numerically. Those greater than 2.0 may be listed as high.

3.4.4.3 Acoustic Levels

The estimated acoustic spectra during flight are presented in Table 3.4.4.4-1.

Table 3.4.4.3-1 Acoustic Levels

Octave Band Center Freq. (Hz)	Sound Pressure Level: db ref. .0002 dynes/cm^2			
	Thor/Delta		Titan III D	Shuttle
31.5	129	124	124	131
63	130	125	130	137
125	134	129	138	141
250	139	134	143	143
500	147	142	142	143
1000	141	136	137	141
2000	138	133	133	138
4000	131	126	130	134
8000	128	123	128	130
Overall	149	144	147	149
Duration (Seconds)	20	70	120	120

3.4.4.4 Shock

TBD.

Table 3.4.4-2. Particulate Radiation

Solar High Energy Particle Radiation

Composition: Predominantly of protons (H^+) and alpha (He^{++})

Integrated yearly flux:

Energy > 30 Mev	N 8×10^9 protons/cm^2 near solar maximum.	
	N 5×10^8 protons/cm^2 near solar minimum.	
Energy > 100 Mev	N 6×10^8 protons/cm^2 near solar maximum.	
	N 1×10^8 protons/cm^2 near solar minimum:	

Maximum dosage with shielding of 5 gm/cm^2: 200 rads per week (3 flares)

Table 3.4.4-3. Meteoroid Environment

The encounter frequency (N), in number per square meter per second, of sporadic cometary meteoroids with a mass equal to or greater than m grams on a randomly oriented surface is

$$\log N = -.05426\ (\log m)^2 - 1.614 \log m$$

$$- 14.644 + \log\ 1 + (0.419/r)$$

$$+ \log\ 1 + \ 1 - 1/r^2)\ /2 \ + \log F_{seasonal}$$

where r is the distance from earth in units of earth's radius (6.378×10^3 km) and $F_{seasonal}$ is a seasonal factor obtained from the table given below. The seasonal factor is obtained by taking the average of monthly factors listed for the months of the mission duration.

Seasonal Factors	
January	.6
February	.4
March	.5
April	.6
May	1.1
June	1.6
July	1.8
August	1.6
September	1.1
October	1.1
November	.9
December	.7

3.4.5 Transportability

The design of the General Purpose Spacecraft and its packaging and packing for shipment shall be such that the spacecraft will meet all performance requirements stated in Section 3 of this specification after the spacecraft in the AGE transportaion equipment is subjected to the transportation environments described in paragraph 3.4.4 of this specification. Transportation shall be by highway and/or air supported in the AGE shipping container on a test and calibration dolly.

Explosive devices shall be shipped separately from the spacecraft for installation at the launch site. These devices shall be packaged and transported in accordance with ICC Tariffs No. 6C (for commercial aircraft) and No. 19 (for other than commercial aircraft.)

3.4.6 Safety

3.4.6.1 Ground Safety

Design consideration shall be given to minimize hazardous interaction of equipment, facilities, and facility equipment during spacecraft manufacture, test, and final installation. Suitable precautions shall be specified in spacecraft handling, assembly, and test instructions. Ground operating procedures shall incorporate warning and cautionary instructions to preclude inadvertent equipment damage or personnel injury resulting from inherent RF radiation, explosive, and pressure vessel hazards. Parts which may work loose in service shall be safety wired in accordance with MS33540, or shall have other approved locking means applied.

3.4.6.2 Personnel Safety

Personnel safety shall be in accordance with MIL-STD-454, Requirement 1. Adequate means for preventing inadvertent deployment of the solar array during ground handling and test operations shall be employed.

3.4.6.3 Explosive and/or Ordnance Safety

Safe-arm plugs shall be used to maintain a short circuit across each ordnance device during assembly and ground handling. Shorting plugs shall be placed in the harness connecting the firing circuit (pyrotechnic controller) to the explosive devices and shall be located in a close proximity to the explosive devices. The shorting plugs shall be replaced by arm plugs during preparation of the spacecraft for launch. The requirements for propellant (hydrazine) and explosive device handling and transfer shall be in accordancne with AFWTRM-127-1 and AFM-71-4.

3.5 DESIGN AND CONSTRUCTION

3.5.1 General Design Features

3.5.1.1 Spacecraft Reference Axes

Spacecraft reference axes for determination of mass properties sahll be as shown in Figure 3.5.1.1-1.

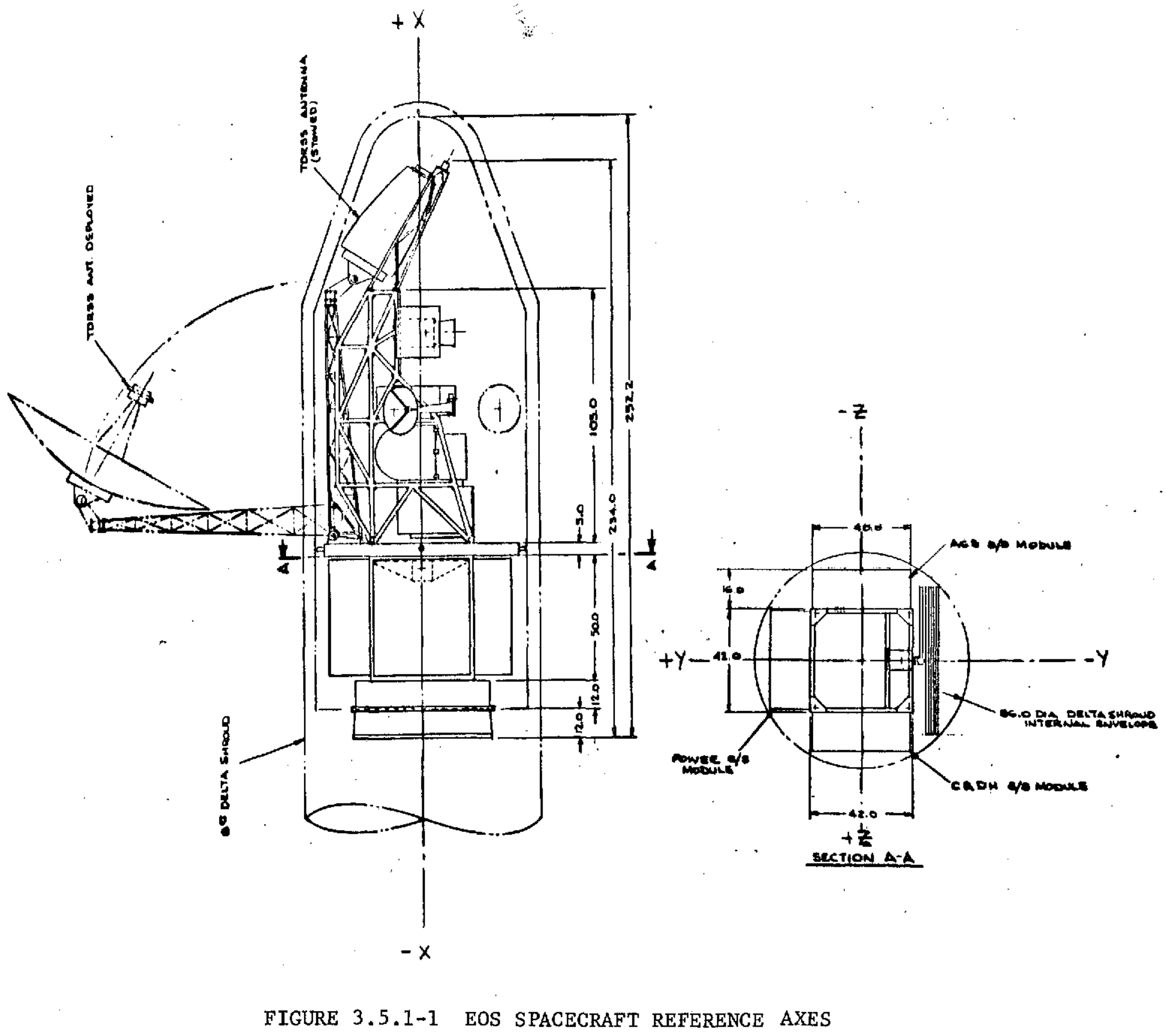


FIGURE 3.5.1-1 EOS SPACECRAFT REFERENCE AXES

3.5.1.2 Mass Property Restraints

3.5.1.2.1 Launch Weight

The General Purpose Spacecraft segment weight, including the structure, ACS, Power, C&DH, harness and signal conditioning, thermal control, and pneumatics shall not exceed 1125 lbs.

3.5.1.2.2 Center of Mass *

The radial offset of the center of mass from the z axis (yaw) shall not exceed 1.0 inches in the separation mode. (Less adapter & array folded).

In the orbit mode (less adapter, array open)the position of the center of mass shall not deviate more than

$x = \pm\ 0.5$ inches

$z = \pm\ 0.5$ inches

$y = \pm\ 0.5$ inches

3.5.1.2.3 Products of Inertia *

In the orbit mode (less adapter, array open) the products of inertia shall not exceed,

Pxz = 0 (spacecraft ballasted)

Pxy = $\pm$ N.A. Sl-Ft.^{2}

Pzy = $\pm$ 3 Sl-Ft.^{2}

3.5.1.2.4 Moments of Inertia *

In the orbit mode (less adpater, array open) the moments of inertia shall not exceed,

Ix = TBD
Iz = TBD
IY = TBD

* The center of mass, products and moments of inertia requirements exist for a total spacecraft, which includes the General Purposė Spacecraft Segment and a Mission Peculiar Spacecraft Segment.

3.5.1.3 General Purpose Spacecraft Segment Configuration

An exploded view of the General Purpose Spacecraft is shown in Figure 3.5.1.3-1. In the case of the Propulsion Module, however, the Reaction Control System is the only section applicable to the GPS/CS.

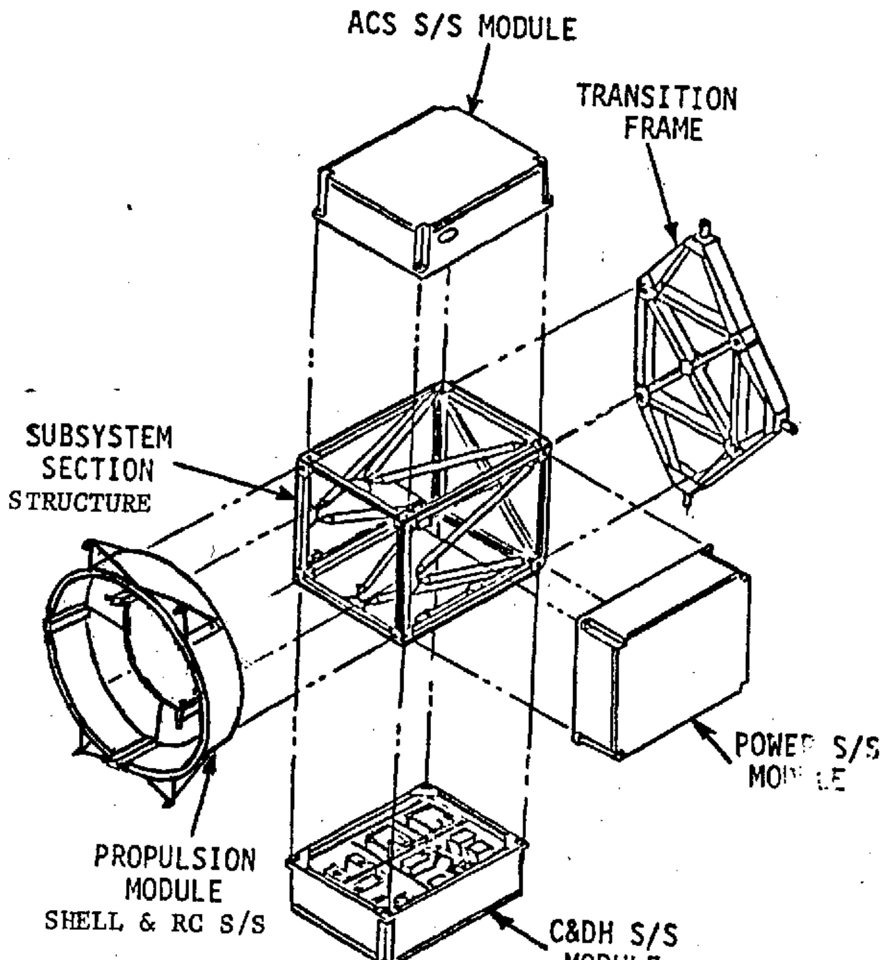


FIGURE 3.5.1.3-1 GENERAL PURPOSE SPACECRAFT SEGMENT EXPLODED VIEW

3.5.1.4 Structures Subsystem *

The General Purpose Spacecraft Segment structure shall provide the primary and secondary structural elements to control the positional relationships of and provide mounting surfaces for the other subsystems. The structure shall possess sufficient strength and rigidity to maintain critical alignments of subsystems during pre-launch, launch and orbital environments.

SVS XXXX, "Specification for the EOS Structures Subsystem" defines in detail the performance and design of the structures and mechanisms required for the General Purpose Spacecraft Segment.

3.5.1.5 Thermal Subsystem

The Thermal Control Subsystem is essentially a passive system which utilizes insulation, coatings and heaters. Thermal control at the Module level will be at the following levels:

ACS	$70^{\circ} \pm 5^{\circ}$ F
C&DH	$70^{\circ} \pm 5^{\circ}$ F
Power	$50^{\circ} \pm 5^{\circ}$ F

The thermal control coating will be teflon/silver.

SVS XXXX, "Specification for EOS Thermal Control System" presents the specific details of performance and design fo this system for the General Purpose Spacecraft Segment.

3.5.1.6 Communications and Data Handling Subsystem Module

The C&DH Subsystem Module provides the means for spacecraft tracking, ground and on-board control of all spacecraft functions and for retrieval of spacecraft status data.

* Paragraphs 3.5.1.4 through 3.5.1.11 refer to subsystem or module specifications which define in detail the requirements of each section. The referenced specifications are part of the overall General Purpose Spacecraft Segment Specification and are considered to be requirements of the General Purpose Spacecraft Segment.

Both USB and TDRSS links are provided for command, ranging, narrowband and medium rate telementry.

SVS XXXX, "Specification for EOS Communications and Data Handling Subsystem Module" defines the details of performance and design of this module for the General Purpose Spacecraft Segment.

3.5.1.7 Basic Software

A set of Basic Software is required to be furnished with the on-board computer. This software will be modular so that mission unique functions may be added to the flight program. The basic programs are an executive which handles I/O and task management, a stored command handler; and a status buffer handler which reports observatory back-orbit status.

The above programs will be developed for the Advanced On-Board Computer (AOP). Various other subsystem programs such as thermal control, power regulator control, momentum unloading, gyro update, etc. will also be required.

SVS XXXX, "Specification for EOS Basic Software" defines the details of the above software for the General Purpose Spacecraft Segment.

3.5.1.8 Attitude Control Subsystem Module

The EOS Attitude Control Subsystem Module is a precise stellar reference subsystem with maximum flexibility.

The sensors for the ACS are a fixed head 2-axis star tracker with a Silicon detector, double degree of freedom gyros for the inertial reference unit and a solar aspect sensor for acquisition.

Maximum use of the AOP for control laws and logic is provided with a modular software structure.

Reaction devices consist of momentum wheels, "coarse" (0.25 lb F) reaction control system and magnetic torquers.

SVS XXX, "Specification for EOS Attitude Control Subsystem Module" defines the details of performance and design of this module for the General Purpose Spacecraft Segment.

3.5.1.9 Reaction Control Subsystem

This specification addresses only the reaction control system portion of the propulsion subsystem which provides the functions of initial stabilization/re-stabilization and back-up momentum unloading.

SVS XXXX, "Specification for the EOS Hydrazine Reaction Control Subsystem" defines the details of performance and design for this section of the General Purpose Spacecraft Segment.

3.5.1.10 Power Subsystem Module

The Power Subsystem Module is designed to permit normal operations of the General Purpose Spacecraft Segment and the mission peculiar payload in conjunction with the Solar Array which is also treated as mission peculiar.

Details of the performance and design of the Power Subsystem Module are presented in SVS XXXX, "Specification for EOS Power Subsystem Module".

3.5.1.11 Electrical Integration Subsystem

The module design concept of the spacecraft requires that the interface between modules be minimized and reliable. Signal distribution is primarily dependent on the command and telemetry party lines. The electrical integration of the spacecraft provides a bus capable of supporting the data flow among the ground, OBC, and spacecraft subsystems.

The requirements for electrical distribution include: (1) Signal distribution compatible with subsystem, and launch vehicle interfaces and extent of OBC control; (2) Electromagnetic compatibility requirements based on MIL-STD-461 and 462; (3) Protection against radiation environment expected for each mission; (4) Power bus protection, and (5) Command and telemetry requirements.

The design provides for two major harness segments, one on each side of the transition ring. This is consistent with the party line concept of signal distribution which minimizes the number of wires in each segment. Rack and panel, blind mate, self-aligning, deadface connectors containing signal, power, and coaxial contact are selected to simplify alignment during mating. Harnesses within each module will be segmented between subassemblies to permit easy replacement.

Details of the performance and design of the Electrical Integration Subsystem are presented in SVS XXXX, "Specification for EOS Electrical Integration Subsystem."

3.5.1.12 Alignment

Critical alignment requirements for General Purpose Spacecraft Segment items are given in Table (later). The requirements are presented for initial alignment (installation prior to environmental testing and for recheck - remeasurement after environmental test). Also defined in the table are the measurement accuracies required. Unless otherwise specified, all alignment requirements specified are with respect to the spacecraft geometric axes.

3.5.2 Selection of Specifications and Standards

All specifications and standards other than those approved for use by NASA shall be approved by the prime contractor prior to use. Specifications and standards shall be selected in accordance with MIL-STD-143.

3.5.3 Materials, Parts and Processes

Particular attention shall be given to the application and use of materials, parts and processes to facilitate interchangeability, stocking and replacement. Materials shall be chosen on the basis of suitability and availability in the United States. Non critical materials shall be used wherever practical when performance, interchangeability or reliability will not be adversely affected or production significantly altered. Parts shall be selected from the GSFC Preferred Parts List, PPL-11. Parts, materials, and processes used in the fabrication of equipment previously accepted by the government shall be acceptable, provided that all of the following are satisfied:

a. Evidence of prior acceptance is submitted to GSFC.

b. Prior application included demonstration of capability in equivalent or more severe environments than specified in paragraph 3.4.4 herein.

c. The selection is approved by GSFC.

3.5.4 Standard and Commercial Parts

Standard or commercial parts may be used in the AGE subsystems consistent with reliability, maintainability and performance. MS or AN parts shall be used where they are suitable to the application. Commercial parts having suitable properties may be used where no appropriate standard part is available. Standard utility parts (e.g.: screws, nuts, bolts, cotter pins) may be used in the spacecraft providing they exhibit suitable properties and their use is approved by GSFC. For AGE, commercial utility parts having suitable properties may be used provided they can be replaced by standard utility parts without alteration and the corresponding standard parts numbers are referenced in the parts list.

3.5.5 Moisture and Fungus Resistance

Component design shall conform to requirement 4 of MIL-STD-454, except, for AGE, paragraph 2 and all references to MIL-STD-810 are deleted. Wherever possible, non-nutrient materials which resist damage from moisture and fungus shall be used. Protective coatings shall not be acceptable as moisture and fungus preventatives for parts which may lose their coating during the normal course of assembly, inspection, maintenance, and testing.

3.5.6 Corrosion of Metal Parts

The use of dissimilar metals, as defined in MS33586, shall be avoided wherever possible. Materials, techniques, and processes shall be selected and employed with regard to heat treatment procedure, corrosion protection, finish and assembly and installation such that sustained or residual surface tensile stress, stress concentrations, and the hazards of stress corrosion, cracking, and hydrogen embrittlement are minimized. Processes and materials for protection against corrosion of metal parts shall be selected from those specified in paragraph 3.5.3, with the exception that cadmium plating shall not be used. Selected finishes shall be compatible with all the requirements specified for each equipment. Materials and surfaces which may be exposed to an effluent shall be selected for compatibility with the effluent insofar as design considerations permit.

3.5.7 Interchangeability and Replaceability

Mechanical and electrical interchangeability shall exist between like assemblies, subassemblies, and replaceable parts, without modification. Interchangeability shall be defined in accordance with requirement 7 of MIL-STD-454.

3.5.8 Workmanship

Workmanship shall be in accordance with requirements 9 and 24 of MIL-STD-454 and with NHB 5300.4 (3A), and GSFC approved quality program plan.

3.5.9 Electromagnetic Interference

Electromagnetic compatibility shall be in accordance with the requirements of Electromagnetic Interference, Electromagnetic Compatibility (EMI/EMC) Control Plan included in Section _____ of this specification.

3.5.10 Identification and Marking

Identification and marking shall be in accordance with the requirements of MIL-STD-129, 130, MIL-E-5400, paragraph 3.1.16, and MIL-STD-1247. Shipping documentation and containers for flight equipment shall be marked "Items for Space Flight Use".

3.5.11 Electrical AGE

Electrical AGE shall be designed, fabricated and tested in accordance with the requirements of specification (later).

3.6 PERFORMANCE ASSURANCE REQUIREMENTS

3.6.1 Reliability Program

The reliability program shall be implemented in accordance with the requirements of NASA Reliability Publication NHB 5300.4 (1A), as defined in the EOS Reliability Program Plan.

3.6.2 Quality Program

The quality assurance program shall be implemented in accordance with the requirements of NASA Quality Publication, NHB 5300.4 (1B), as defined in the EOS Quality Program Plan.

3.6.3 Test Program

The test program shall be performed in accordance with the provisions of Section 4 of this specification (to be added later). Monitoring and control shall be in accordance with the provisions of the EOS Quality Program Plan.

3.6.4 Configuration Management

Configuration Control shall be maintained in accordance with the provisions of the EOS Configuration Management Plan.

3.6.5 Malfunction Reporting

Malfunctions shall be reported in accordance with the requirements defined in the EOS Failure Analysis and Reporting Plan.

3.6.6 Electrical Connections

Soldered electrical connections shall be made in accordance with the provisions of NASA Document NHB 5300.4 (3A), Requirements for Soldered Electrical Connections.

Specification No. SVS-XXXX
16 September 1974

SECTION 3

SPECIFICATION

FOR THE

EARTH OBSERVATORY SATELLITE (EOS)

STRUCTURES SUBSYSTEM

General Electric Company
Space Systems Organization
Valley Forge Space Center - P.O. Box 8555
Philadelphia, Pa. 19101

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SECTION 1

SCOPE

This specification establishes the performance, design, development, and test requirements for a set of structural subassemblies and mechanisms assembled into a:

a) General Purpose Spacecraft Segment (GPSS), and a

b) Mission Purpose Spacecraft Segment (MPSS)

Together, they form an integrated Structures Subsystem for the Earth Observatory Satellite (EOS).

The GPSS is described primarily in paragraphs 3.1.1.1 and 3.2.2.2; the MPSS is described primarily in paragraphs 3.1.1.2 and 3.2.2.3.

SECTION 2

APPLICABLE DOCUMENTS

The following documents of the exact issue shown form a part of the specification to the extent referenced herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification will be considered the superseding requirement.

2.1 GENERAL ELECTRIC DOCUMENTS

TBD	General Purpose Spacecraft Specification
TBD	Mission Peculiar Spacecraft Specification
TBD	EOS/Delta 2910 Interface Specification
TBD	EOS/Titan III B Interface Specification
TBD	EOS/Space Shuttle FSS Interface Specification
TBD	Approved Materials and Processes List
TBD	Approved Parts List
TBD	Thermal Control Subsystem (TCS) Specification
TBD(s)	Aerospace Ground Equipment (AGE) Specifications
TBD(s)	Auxiliary Aerospace Equipment (AAC) Specifications
TBD	Electromagnetic Compatibility
TBD	EOS/Delta 2910 Interface Drawing
TBD	EOS/Titan III B Interface Drawing
TBD	EOS/Space Shuttle FSS Interface Drawing
TBD	Launch Vehicle Adapter/Spacecraft Separation Drawing
TBD	Secondary Propulsion Subsystem (SPS) Component Location and Space Envelope
TBD	Structures Subsystem Performance Requirements
TBD	GPSS Structure Assembly

TBD	Solar Array Assembly and Drive Installation
TBD	Solar Array Panel Assembly
TBD	Handling Provisions Documents/Drawings
TBD	Launch Vehicle Separation Subsystem Specification
TBD	Instrument Section Specification
TBD	Tracking and Data Handling Antenna Specification
TBD	Wideband Antenna Specification
TBD	Fixed Module Latching Mechanism Specifications
TBD	Fixed Module Latching Mechanism Drawings
TBD	Replaceable Module Latching Mechanism Specification
TBD	Replaceable Module Latching Mechanism Drawings
TBD	Electrical Harness Installation
TBD	GPSS/MPSS Interface Specification
TBD	Component Accessibility Drawing
TBD	GPSS/MPSS Interface Drawing
TBD	Sensor Fields of View

2.2 U.S. GOVERNMENT DOCUMENTS

The following documents of the exact issue shown or, if not shown, of the issue in effect on the date of the request for proposal, form a part of this specification to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

2.2.1 MILITARY SPECIFICATIONS

MIL-E-8983A 30 November 1971	Electronic Equipment, Aerospace Extended Environment, General Specification for
MIL-Q-9858A 16 December 1963	Quality Program Requirements

MIL-W-8604 Amendment 1 8 October 1965	Welding of Aluminum Alloys, Process for
MIL-D-1000 1 March 1965	Drawings, Engineering and Associated Lists
MIL-W-6858C Amendment 1 28 June 1965	Welding, Resistance; Aluminum, Magnesium, Non-hardening Steels or Alloys, Nickel Alloys, Heat Resistance Alloys and Titanium Alloys, Spot and Seam
MIL-W-8160D Amendment 1 24 December 1963	Wiring, Guided Missile, Installation of, General Specification for
MIL-I-8500B 10 October 1960	Interchangeability and Replaceability of Component Parts for Aircraft and Missiles
MIL-B-7883B 20 February 1968	Brazing of Steels, Copper, Copper Alloys, Nickel Alloys, Aluminum and Aluminum Alloys

2.2.2 NASA SPECIFICATIONS

NASA-S-320-G-1 October 1969	General Environmental Test Specification for Spacecraft and Components

2.2.3 FEDERAL STANDARDS

FED-STD-209A 10 August 1966	Clean Room and Work Station Requirements, Controlled Environment

2.2.4 MILITARY STANDARDS

MIL-STD-454C Notice 2 1 December 1971	Standard General Requirements for Electronic Equipment
MIL-STD-889 25 September 1969	Dissimilar Metals
MS 33540F 12 August 1969	Safety Wiring, General Practices for
MIL-STD-143B 12 November 1969	Specifications and Standards, Order of Precedence for Selection of
MIL-STD-130D Change 1 30 July 1971	Identification and Marking of U.S. Military Property

MIL-STD-1472A 15 May 1970	Human Engineering Design Criteria for Military Systems Equipment and Facilities
MIL-STD-100A 1 October 1967	Engineering Drawing Practices
MIL-STD-810B September 1969	Environmental Test Methods for Aerospace and Ground Equipment

2.2.5 REGULATIONS

Tariff #19	Interstate Commerce Commission Regulations for Transport of Explosives and Other Dangerous Articles (by other than Commercial Aircraft)
Tariff #6C	Interstate Commerce Commission Regulations for Transport of Explosives and Other Dangerous Articles (by Commercial Aircraft)

2.2.6 REPORTS

LV-122-73/B&P-73-41499-12 October 1973	Titan Candidate Launch Vehicles for EOS Missions At WTR
MCR-68-61	Titan III B Users Handbook
DAC-61687 Rev. August 1972	Delta Spacecraft Design Restraints
JSC-07700 Vol. XIV, Rev. B Dec. 21, 1973	Space Shuttle System Payload Accommodations

2.2.7 HANDBOOKS

MIL-HDBK-5B 1 September 1971	Metallic Materials and Elements for Aerospace Vehicle Structure
MIL-HDBK-17 Part 1 1 January 1971	Plastics for Aerospace Vehicles; Reinforced Plastics
MIL-HDBK-17 Part 2 14 August 1961	Plastics for Flight Vehicles; Transparent Glazing Materials
MIL-HDBK-23A 30 December 1968	Structural Sandwich Composites

SECTION 3

REQUIREMENTS

3.1 SUBSYSTEM DEFINITION

3.1.1 SUBSYSTEM DESCRIPTION

The Structures Subsystem forms the basic structural framework of a modular-concept Earth Observatory Satellite (EOS). The subsystem provides the appropriate component mounting surfaces and mechanisms to: (1) integrate the General Purpose and the Mission Peculiar Spacecraft Segments of the EOS; (2) interface with, and separate from the launch vehicle; (3) deploy and orient the solar array panels and antenna assemblies; and (4) permit retrieval by the Space Shuttle.

The orbit configuration of the EOS is illustrated in Figure 3.1-1; the modular configuration is shown in 3.1-2.

3.1.1.1 General Purpose Spacecraft Segment (GPSS)

The General Purpose Spacecraft Segment includes:

- o Subsystem Section Structure
- o Subsystem Module Frames, for:
  - -- Electrical Power (EPS)
  - -- Attitude Control (ACS)
  - -- Communications and Data Handling (C&DH)
- o Subsystem Module Attachment Mechanisms
- o Solar Array Retention Mechanism
- o Transition Frame Assembly
- o Propulsion Module Shell, and R/C Thrusters

The general arrangement of the General Purpose Spacecraft Segment, in relationship to the Mission Peculiar Spacecraft Segment, is illustrated in Figure 3.1-3.

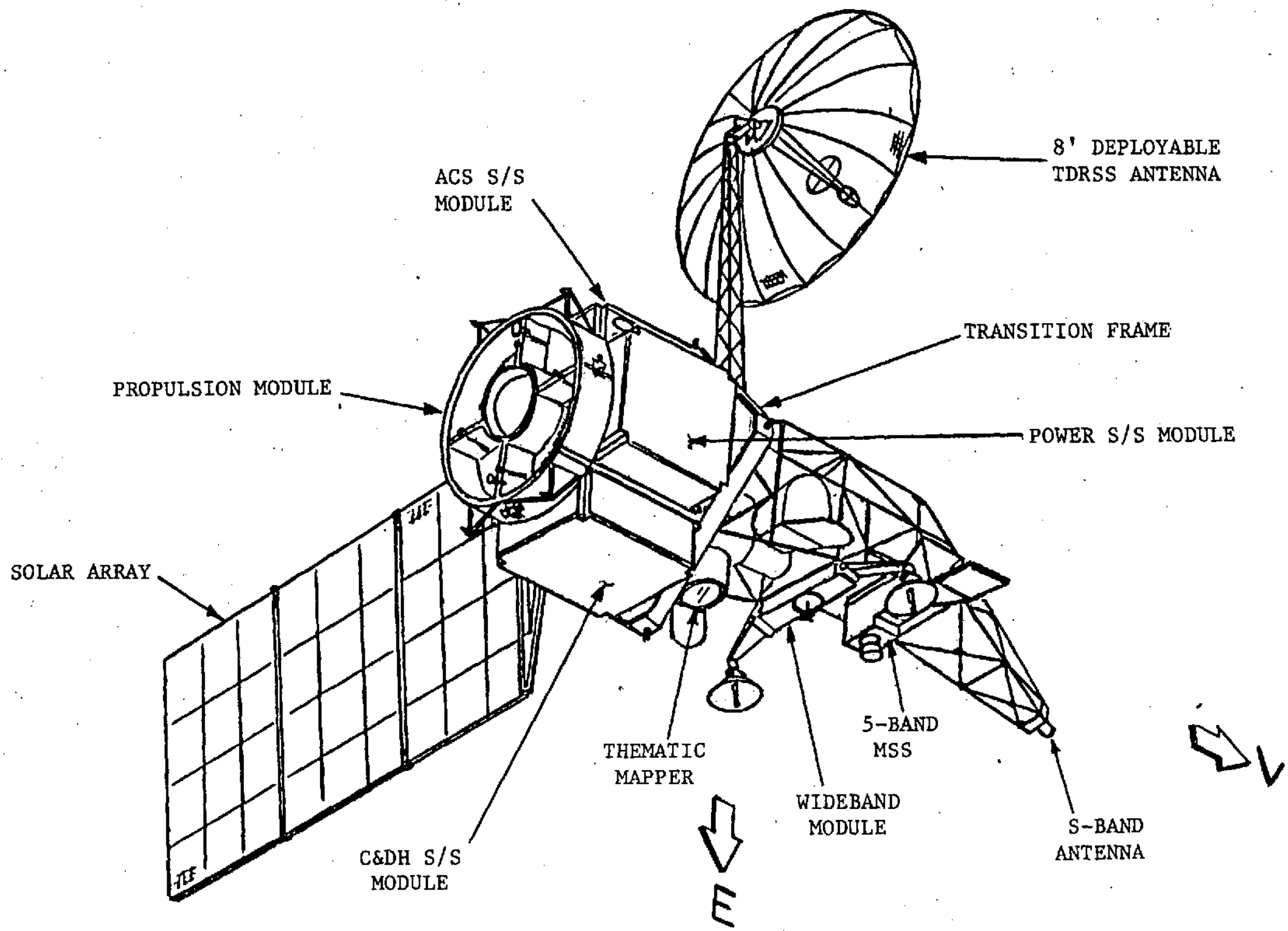

Figure 3.1-1. EOS Orbital Configuration

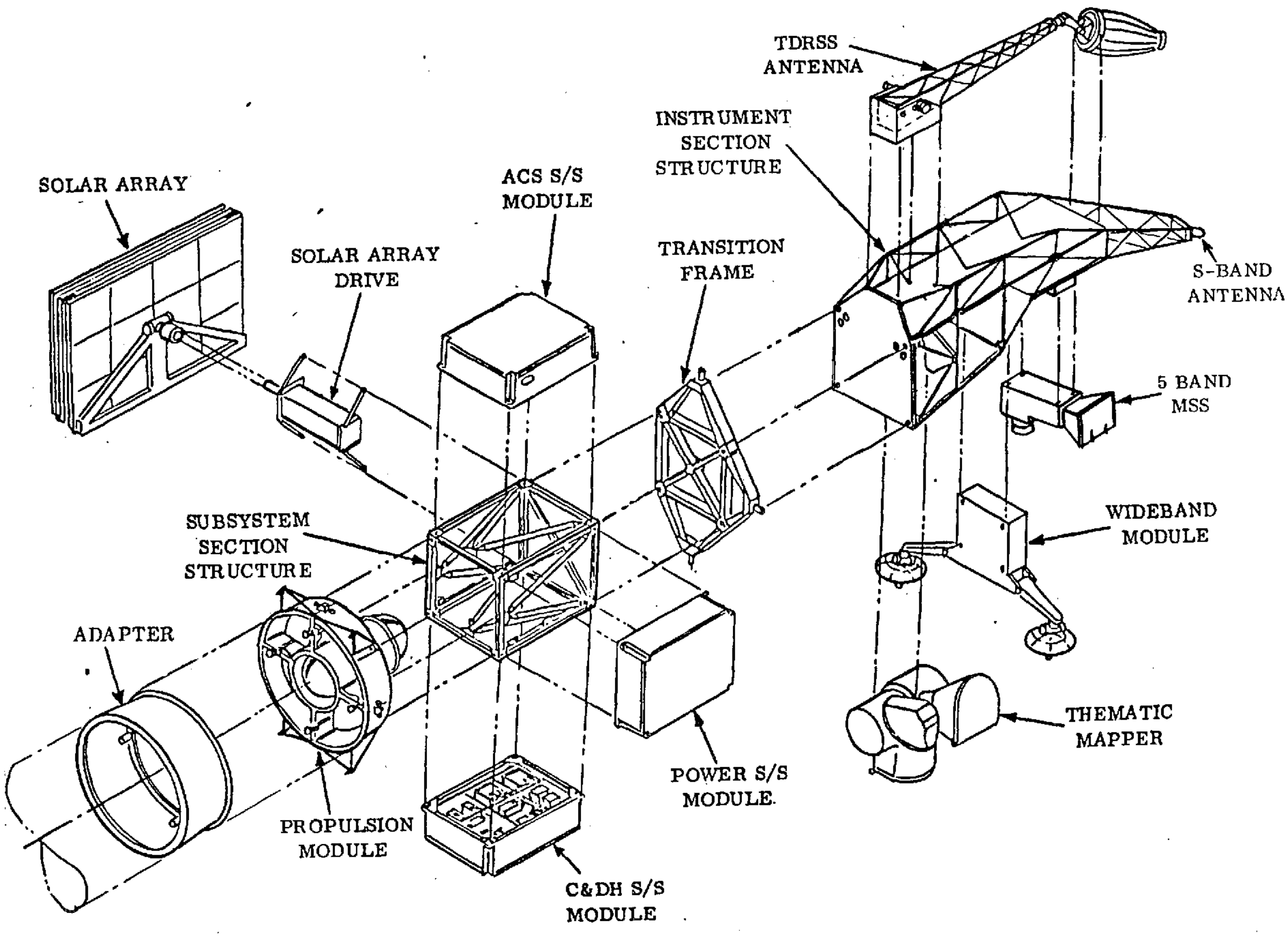

Figure 3.1-2 EOS Modular Configuration

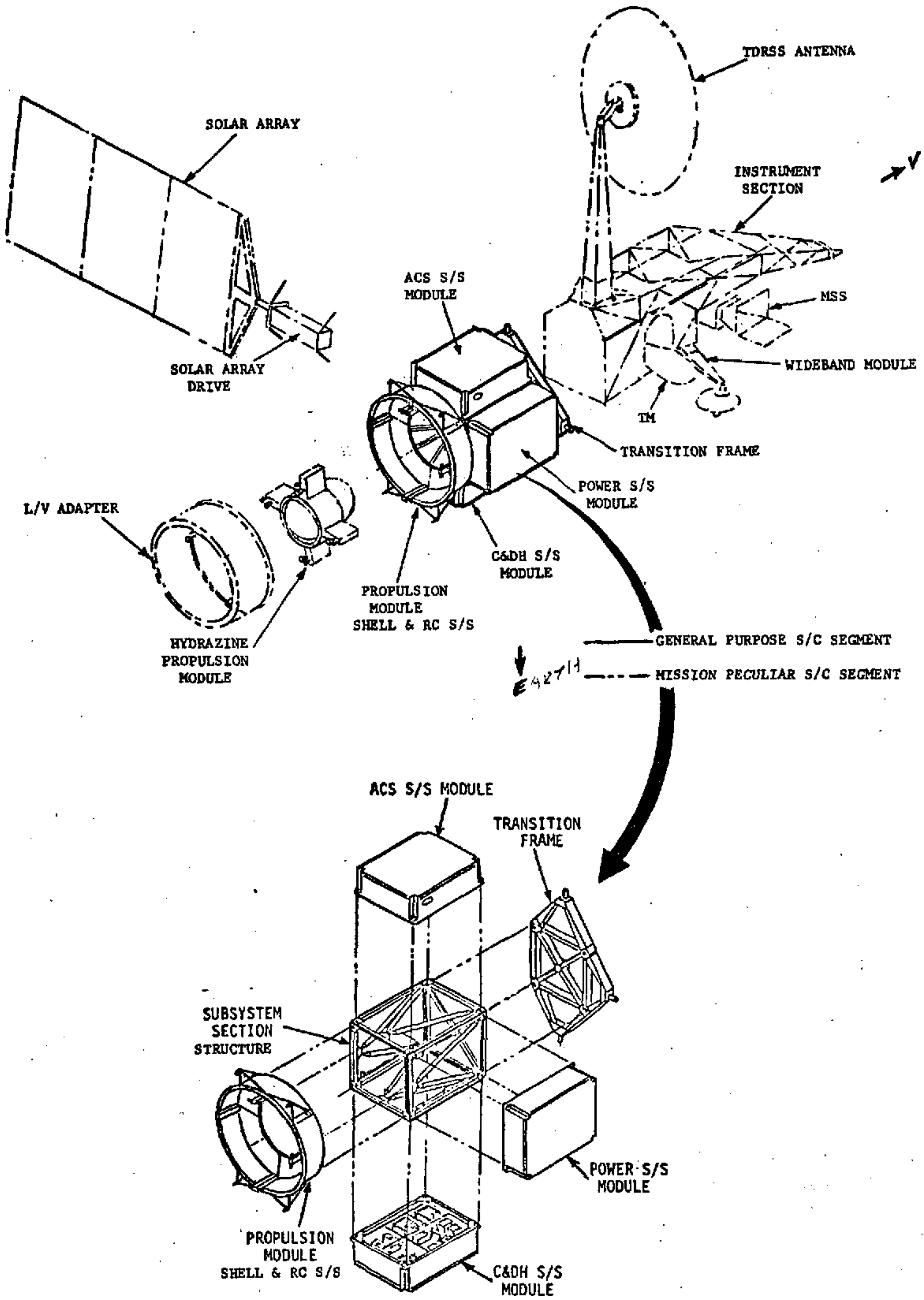

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Figure 3.1-3. General Purpose Spacecraft Segment

3.1.1.2 Mission Peculiar Spacecraft Segment (MPSS)

The Mission Peculiar Spacecraft Segment includes:

- o Launch Vehicle Adapter
- o Hydrazine Propulsion Module
- o Instrument Section Structure
- o Launch Vehicle Separation Mechanism
- o Solar Array Assembly
- o Solar Array Deployment/Retraction Mechanism
- o Solar Array Drive Assembly
- o Antenna Deployment/Retraction Mechanism

The general arrangement of the Mission Peculiar Spacecraft Segment (MPSS) in relationship to the GPSS, is shown in Figure 3.1-4.

3.1.2 INTERFACE DEFINITION

The Structure Subsystem shall be designed to be compatible with the external interface requirements such as the:

- o Launch Vehicle(s)
- o Space Shuttle Flight Support System
- o Aerospace Ground Equipment (AGE)
- o Auxiliary Aerospace Equipment (AAE)

plus the internal interface requirements such as the:

- o GPSS/MPSS integration
- o Attitude Control Subsystem
- o Electrical Power Subsystem
- o Communications and Data Handling Subsystem (C&DH)
- o Thermal Control Subsystem, and the
- o Mission Payload/MPSS Integration.

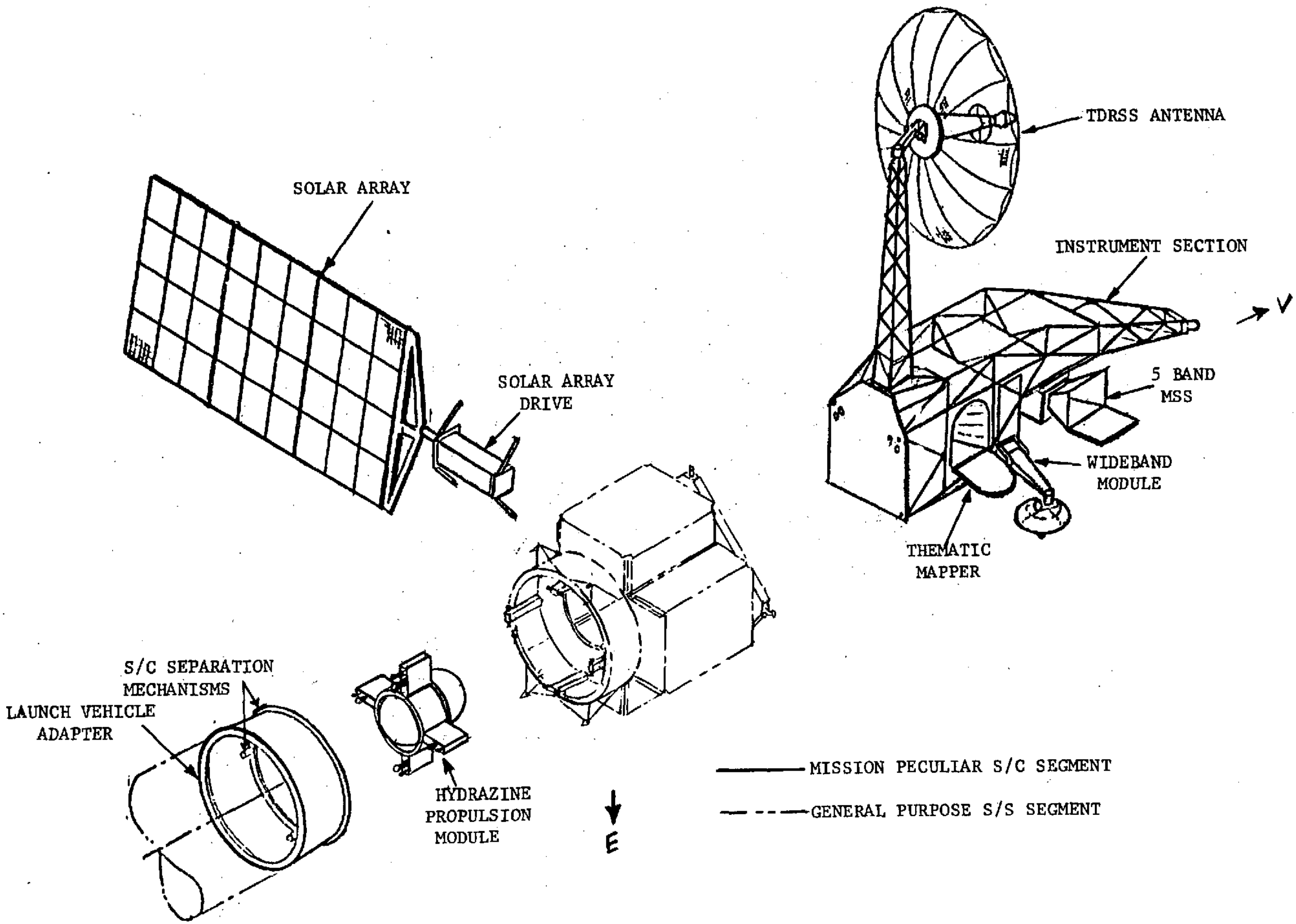


Figure 3.1-4. Mission Peculiar Spacecraft Segment

3.1.2.1 GPSS/MPSS Interfaces

The matching surfaces of the General Purpose Spacecraft Segment and the Mission Peculiar Spacecraft Segment shall be compatible with the Interface Document (TBD) and Drawing (TBD).

3.1.2.2 Launch Vehicle Interfaces

The EOS structure subsystem shall be capable of interfacing with the Delta 2910 and Titan III B/NUS for launching per Interface Document (TBD) and Interface Drawings (TBD).

3.1.2.2.1 Delta 2910 Interface

The MPSS shall mechanically interface with the Delta 2910 Launch Vehicle at the launch vehicle adapter mounting plane Station 14.0 in accordance with General Electric Specification TBD (ICD) and Drawing TBD. Provisions shall be made by the MPSS for separation event monitoring through the EPS on the spacecraft side of the separation plane.

3.1.2.2.2 Titan III B/NUS Interface

The MPSS shall mechanically interface with the Titan IIIB/NUS launch vehicle at the launch vehicle adapter mounting plane Station TBD in accordance with General Electric Specification TBD (ICD) and Drawing TBD. Provisions shall be made by the MPSS for separation event monitoring through the EPS on the spacecraft side of the separation plane.

3.1.2.3 Space Shuttle Interfaces

The EOS structure subsystem shall be compatible with the Flight Support System and the Retrieval Systems of the Space Shuttle per Interface Document (TBD) and Interface Drawings (TBD).

3.1.2.3.1 Transition Frame

The GPSS Transition Frame shall mechanically interface with the Shuttle Flight Support System (FSS) Retention Cradle in accordance with the General Electric Specification TBD (ICD) and Drawing TBD.

3.1.2.3.2 Handling Provisions

Provisions shall be made on the Structure Subsystem, in accordance with GE Document (TBD) and Drawing (TBD), for handling, docking, and servicing by the Space Shuttle Retrieval System and Module Exchange Mechanisms.

3.1.2.4 Mechanical Interfaces

3.1.2.4.1 Launch Vehicle Adapter/Propulsion Module Shell Assembly

The Launch Vehicle adapter structure shall support and mechanically interface with the Secondary Propulsion Shell Assembly via Vee-Band Retention/Spring Separation Device in accordance with Drawing TBD.

3.1.2.4.2 Secondary Propulsion Subsystem (SPS)

The GPSS shall provide a modular shell assembly for mounting and supporting of the SPS components. The space envelope for location of these components shall be in accordance with Drawing TBD.

3.1.2.4.3 Secondary Propulsion Shell Assembly/Subsystem Section Structure

The Secondary Propulsion Subsystem Shell Assembly shall support and mechanically interface with the Subsystem Section Structure in accordance with GPSS/MPSS interface document (TBD) and drawing (TBD).

3.1.2.4.4 Subsystem Section Structure/Subsystem Module Frame

The Subsystem Section Structure of the GPSS shall support and mechanically interface with the three Subsystem Module Frame Assemblies in accordance with Drawing TBD for fixed installations or with Drawing TBD for replaceable subsystem modules.

3.1.2.4.5 Subsystem Section Structure/Transition Frame Assembly

The Subsystem Structure of the GPSS shall support and mechanically interface with the Transition Frame in accordance with GE drawing TBD.

3.1.2.4.6 Transition Frame/Instrument Section Structure

The Transition Frame of the GPSS shall support and mechanically interface with the Instrument Section Structure of the MPSS in accordance with GPSS/MPSS Interface Document (TBD) and Drawing (TBD).

3.1.2.4.7 Instrument Section Structure/Mission Peculiar Payloads

The Instrument Section Structure of the MPSS shall support and mechanically interface with the Mission Peculiar Payloads in accordance with GE Document TBD, Instrument Section Specification.

3.1.2.4.8 Subsystem Section/Solar Array Support Structure

The Subsystem Section Structure of the GPSS shall support and mechanically interface with the Solar Array Support Structure and Drive Assembly in accordance with Drawing TBD.

3.1.2.4.9 Solar Array Support Structure

The solar array support structure shall mechanically interface with the EPS subsystem solar array panel assemblies and the drive assembly as shown on Drawing TBD.

3.1.2.4.10 Component Installations

The Structure Subsystem shall provide for installation, orientation, and mounting provisions of all subsystem equipments housed within the Subsystem Section Structure and Subsystem Module Frame assemblies within the specified tolerances and Fields of View shown on Drawing TBD.

Components mounted within the Structure Subsystem shall conform to the thermal interface constraints of paragraph 3.1.2.6.1. Component design shall include provisions for not less than four attachments for structural mounting. Sizing of attachments shall not be less than one-quarter inch diameter for bolted connections nor three-sixteenths diameter for screw type fasteners. Design of the components mounting base shall include edge distance provisions for the next larger size attachment.

3.1.2.4.11 Electrical Harness

The Structure Subsystem shall provide for routing and mounting of the primary spacecraft electrical harness within the space envelope and constraint shown on Drawing TBD.

3.1.2.4.12 Thermal Blankets

The Structure Subsystem shall provide structural mounting and installation of thermal control subsystem insulation blankets within the space allocation and arrangement shown in Drawing TBD.

3.1.2.5 Electrical Interfaces

The electrically functional components of the Structure Subsystem shall be designed to conform with the electrical interface characteristics specified in TBD. In addition, the Structure Subsystem will provide the following electrical interfaces with the electrical power subsystem (EPS).

3.1.2.5.1 Electro Explosive Devices (EED's)

Subsystem EED's shall have a bridgewire resistance equivalent to 1.1 $\pm$ 0.1 ohms with a minimum all-fire current of 3.5 amperes for 10 milliseconds and maximum no-fire current equivalent to one ampere or one watt for 5 minutes duration. The EPS shall provide simultaneity between signals to any EED in sequence within a tolerance of TBD milliseconds.

3.1.2.5.2 Verification Switches

Redundant switches shall be provided as part of the Structure Subsystem to verify spacecraft separation, and solar array panel assembly release and lock. The switches shall be capable of carrying a current of TBD amperes. Double contacts shall be provided on each separation switch to permit monitoring by spacecraft telemetry. The separation switches shall be wired in a quad redundant configuration by the EPS.

3.1.2.6 Thermal Interfaces

3.1.2.6.1 Insulation, Coatings and Finishes

The Structure Subsystem shall provide for thermal insulation, coatings and finishes to component mounting interfaces, as shown on Drawing TBD. Coatings and finishes for the spacecraft shall conform to the optical property characteristics shown in Table 3.1-1 and shall be selected in accordance with the Approved Parts List (TBD) and Approved Materials and Processes List (TBD).

3.1.2.6.2 Conductance

The overall contact conductance between heat generating component base plates and mounting structure shall be maintained by the Structure Subsystem to not less than 200 BTU/hrs-ft^2-^oF. Other components shall have a maximum thermal conductance between the component and structure as specified in SVS-TBD. Thermally conductive compounds shall be selected from (TBD). The EPS component mounting panel shall have a minimum transverse and lateral thermal conductivity of TBD and TBD BTU/hrs-ft^2-^oF; the ACS component mounting panel shall have a minimum transverse and lateral thermal conductivity of TBD and TBD BTU/hrs-ft^2-^oF; the C&DH component mounting panel shall have a minimum transverse and lateral thermal conductivity of TBD and TBD BTU/hrs-ft^2-^oF, respectively. The conductance through the subsystem module attachment mechanisms shall not be greater than TBD BTU/hr - ^oF.

3.1.2.6.3 Surface Flatness

Component baseplate mounting surfaces shall maintain the maximum flatness tolerances and RMS surface roughness characteristics specified in the Thermal Control Subsystem Specification (TBD).

3.1.2.6.4 Alignment

Alignment of the star sensor to the Inertial Reference Unit (ref. Table 3.2-9) shall be maintained by limiting the thermal gradient between the TBD of the star sensor and the TBD of the IRU to less than TBD ^oF.

Table 3.1-1. Thermal Coating Optical Properties

(for Energy Wavelength of Between 0 to 47 Microns)

Structural Element	Absorptivity	Hemispherical Emissivity
LAUNCH VEHICLE ADAPTER		
o Internal Faces	NR	.1 Max
o External Surface	NR	.1 Max
PROPULSION MODULE STRUCTURE		
o Internal Faces	NR	.1 Max
o External Surfaces	NR	.1 Max
o Tank Mounting	NR	.1 Max
o Thruster Mounting	NR	.1 Max
SUBSYSTEM SECTION TRUSS		
o Tubes	NR	.1 Max
o Module Mounting	NR	.1 Max
o Solar Array Mounting	NR	.1 Max
o Transition Frame Mounting	NR	.1 Max
SUBSYSTEM MODULE STRUCTURES (3)		
o Module Mounting Face	NR	.9 Min
o Component Mounting (*)	NR	.9 Min
o Other Interior Faces	NR	.9 Min
o External Faces	NR	.9 Min
TRANSITION FRAME		
o Lateral Faces	NR	.1 Max
o External Faces	NR	.1 Max
o Mounting Faces	NR	.1 Max
INSTRUMENT SECTION		
o Component Mounting (*)	NR	.9 Max
o Other Internal Faces	NR	.9 Max
o External Surface	NR	.9 Max
*Contact surfaces to be alodine or anodize only.		

3.1.2.7 Aerospace Ground Equipment (AGE) Interfaces

The following associated AGE and Auxiliary Aerospace Equipment (AAE) shall be designed to mechanically interface with the Structure Subsystem.

TBD

3.2 CHARACTERISTICS

3.2.1 PERFORMANCE

3.2.1.1 Functional Performance

The Structure Subsystem designed in accordance with the performance requirements of (TBD) shall provide the primary and secondary structural elements and mechanism components required to perform the functions of both the GPSS and MPSS.

3.2.1.1.1 GPSS Functions

The General Purpose Spacecraft Segment functions include:

- Provide the structural base for integration of the Mission Peculiar Spacecraft Segment Modules with the GPSS modules.
- Transfer of launch loads through the GPSS to the MPSS subsystem.
- Mounting, orientation, and control of the positional relationships of the MPSS subsystems.
- Mounting, orientation, and control of the positional relationships of the components within the Attitude Control, Electrical Power, and the Command and Data Handling Subsystems.
- Contribute to spacecraft structural integrity and rigidity during ground operations, launch, and in orbit.
- Alignment control of the Attitude Control Subsystem.
- Retention and release of the solar array panels.
- Incorporation of provisions for servicing and/or retrieval by the Space Shuttle.

3.2.1.1.2 MPSS Functions

The Mission Peculiar Spacecraft Segment functions include:

- o Mounting, orientation, and control of the potential relationship of the components associated with the Mission Peculiar subsystems, i.e., Instrument Section, Solar Array Assembly and Drive Mechanisms, Secondary Propulsion, and the Launch Vehicle Adapter.
- o Alignment control of the payload sensors and the Instrument Section Module.
- o Transfer of launch loads between the spacecraft and the Delta 2910 (or Titan III B/NUS) launch vehicle.
- o Separation of EOS from the launch vehicle.
- o Contribute to the spacecraft structural integrity and rigidity during ground handling, launch, and on orbit.
- o Deployment and retraction of the solar array assembly.
- o Sun tracking of and power take-off from the solar array.
- o Deployment and retraction of the Wide Band and the Tracking and Data Relay antenna.

3.2.1.2 Structural Performance

3.2.1.2.1 Support

The Structures subsystem as an integral part of the spacecraft shall be capable of supporting a gross weight, above the launch vehicle interface plane, of 2582 pounds with the weight breakdown as shown in Table 3.2-1 for a reference EOS satellite equipped with a Thematic Mapper (TM), one Multispectral scanner (MSS), and a Tracking and Data Relay Antenna (TDRSS). The basic structure shall be capable of supporting a spacecraft weighing 3500 pounds.

EOS BASELINE CONFIGURATION WEIGHT BREAKDOWN (POUNDS)

Basic Spacecraft		(1115)
Structure & Modules	360	
Attitude Control	90	
Power	222	
Communications & Data Handling	184	
Harness & Signal Conditioning	110	
Thermal	38	
Pneumatics	40	
Adapter	71	
Total Mission Peculiar		(762)
Structure	185	
Solar Array & Drive	114	
Harness & P/L Remotes	35	
Thermal	29	
Orbit Adjust	45	
Orbit Transfer	145	
Wideband Comm.	134	
TDRSS	75	
Payload		(505)
Thematic Mapper	350	
MSS	155	
Weight Contingency		(200)
TOTAL SPACECRAFT		2582 (1171 Kg)

3.2.1.2.2 Stiffness

A. Powered Flight

The primary and secondary structures shall provice adequate stiffness to satisfy the minimum resonant frequency requirements in Table 3.2-2. For this evaluation, the spacecraft and modules shall be analyzed in the lanuch configuration cantilevered from their attachment points.

B. Orbit

In the orbital configuration, a TBD Hertz minimum resonant frequency for appendages shall be adequate to preclude dynamic interaction between the structure and the Attitude Control System. Analytical evaluations of the deployed solar arrays and antennas, including in detail the effects of hinge and orientation mechanism flexibilities, shall be used to demonstrate compliance with this requirement.

3.2.1.2.3 Strength

A. Critical Loading Conditions

For preliminary sizing, the primary structure shall be designed to the qualification level quasi steady-state accelerations of Table 3.2-3. Off-loading conditions for the Delta launch vehicle configuration shall be considered in the analysis. Subsequent dynamic analyses shall determine responses to the qualification vibration test levels of Tables 3.2-4 and 3.2-5, including estimated "notching" levels to prevent excessive dynamic test loads. In these response analyses, a model damping ratio of $C/C_c = 0.05$ will be used for the primary structure modes.

Table 3.2-2. Spacecraft Minimum Resonant Frequencies During Powered Flight

A. Primary Structure

Launch Vehicle / Structure	THOR/DELTA Lat. (Hz)	THOR/DELTA Long. (Hz)	TITAN III B Lat. (Hz)	TITAN III B Long. (Hz)	SHUTTLE Lat. (Hz)	SHUTTLE Long. (Hz)
Spacecraft*	10	30	10	30	TBD	
Subsystem Module**	60	60	60	60	60	60
Experiment***	TBD		TBD		TBD	

*Cantilevered from the launch vehicle/adapter interface, or as supported by by Shuttle FSS.

**Mounted on ball joints at the four attachment points.

***Cantilevered from the transition frame attachment points.

B. Secondary Structure

Item	Longitudinal (Hz)	Lateral (Hz)
Instrument Section Structure	70	70
Antenna Mounting	70	70
Stowed Solar Array Module Mounting	25-30	8
Stowed Solar Array Panel	70	15
Subsystem Component Mounting	100	100

Table 3.2-3. Qualification Level Quasi-Steady Accelerations

Launch Vehicle & Condition	Longitudinal (9)	Lateral (9)
Delta 2910		
Max. Lateral (Lift-off)	- 4.4	±3.0
Max. Compression (MECO/POGO)	-18.0	±1.0
Max. Tension	1.5	±3.0
Titan III B/NUS		
Max. Lateral (Lift-off)	- 2.9	±2.5
Max. Compression (Stage II shutdown)	-13.5	±1.3
Max. Tension (Stage I shutdown)	3.1	±1.9
Shuttle		
Lift-Off	- 3.5	±1.3
Orbiter end burn	- 5.0	±0.6
Entry	0.4	4.5
Landing & Braking	± 2.3	3.8
Crash (ultimate, applied separately)	9.0	4.5
	- 1.5	-2.0

Table 3.2-4. Sinusoidal Vibration

Launch Vehicle / Axis	Thor/Delta		Titan III B		Space Shuttle	
	Frequency (Hz)	Acceleration (G, o-pk)	Frequency (Hz)	Acceleration (G, o-pk)	Frequency (Hz)	Acceleration (G, o-pk)
Longitudinal Axis	5-15* 15-21 21-100	2.3 6.0 2.3	TBD	TBD	TBD	TBD
Lateral Axes	5-14* 14-100	2.0 1.5	TBD	TBD	TBD	TBD

Sweep Rate: 2 octaves/minute.

*Limited with the performance of the exciter. The amplitude in these frequency ranges shall not exceed 0.5 inches D.A.

Table 3.2-5. Random Vibrations

	Frequency (Hz)	PSD (G^2/Hz)	GRMS	TIME (Seconds per Axis)
Thor Delta	20-300 300-700 700-2000	+4 dB/Oct .16 -3 dB/Oct	14.1	20
	20-300 300-700 700-2000	+4 dB/Oct .07 -3 dB/Oct	9.5	70
Titan III B	TBD	TBD	TBD	TBD
Shuttle	20-100 100-250 250-2000	+6 dB/Oct .65 -6 dB/Oct	24.3	90

*Thrust and Lateral Axes.

The final design loads shall be determined in the subsequent phase from coupled launch vehicle-spacecraft dynamic response analysis and dynamic analyses of other critical conditions. These analysis results shall be used to determine the notching factor for the sinusoidal vibration tests at the primary spacecraft resonances such that the primary structure stress levels experienced during flight shall be compatible with those experienced during the vibration test.

B. Shock

Design criteria for shock loads shall be -(TBD).

C. Ground Handling, Transportation and Storage

The structural design shall include consideration of all environments to which the structure and its component parts are exposed during manufacture, ground handling, transportation and storage. Except for ground handling attachments and fittings, the ground loads shall govern design of the structure to the minimum extent practicable.

Appropriate hoisting and support locations may be designed into the basic vehicle structure to facilitate handling and transportation conditions. For additional details of pre-launch environments, MIL-STD-810 is deemed applicable to the EOS Program.

D. Steady State Accelerations

	Nz	Nx	Ny
a. Hoist and Spacecraft to L/V mating	-3.0	±0.5	±0.5

b. Transportation (Rail, Air, Motor, and Water

Vertical: 2.0 g up
4.0 g down
Lateral (sideways): ±2.5g
Longitudinal ±3.0 g (due to docking ramp impact)

These loads are maximum expected equivalent static loads due to carrier operation. They are to be applied separately. The directional terms are with respect to the carrier motion. These loads are to be reacted by the appropriate transportation support configuration. Special procedures, handling equipment, transportation support configuration and shock and vibration isolation between spacecraft and carrier floor will be utilized in order not to exceed the above loads and that the spacecraft element loadings do not exceed 50% of the flight qualification loads.

E. Factors of Safety

The design load factors of safety shown in Table 3.2-6 shall be applied to qualification loads presented in Table 3.2-3 to obtain the structural design yield and the design ultimate loads.

The pressure vessel factors shown in Table 3.2-7 shall be applied to maximum expected operating pressures to obtain design pressures for all hydraulic and pneumatic components.

F. Margins of Safety

The Structure Subsystem when subjected to the environments of paragraph 3.2.5 of the quality assurance provisions of Section 4.0, (TBD), and the design factor of safety of paragraph 3.2.1.2.3(E) shall maintain the minimum design margins of safety presented in Table 3.1-8.

Table 3.2-6. Design Load Factors of Safety

Load Condition	Design Load Factors of Safety	
	Yield	Ultimate
Launch (qualification level)	1.5	2.0
Orbital (qualification level)	1.5	2.0
System Qualification Test	1.5	2.0
Transportation,Handling (Apply Load Factors to loads of paragraph 3.2.1.2.3(A)	1.5	2.0

Table 3.2-7. Pressure Vessel Factors

Pressure Container	Operating	Proof	Burst
Main Propellant Tanks	1.00	1.50	2.00
Vessels Including Accumulators & Pressurization Bottles	1.00	1.50	2.50
Hydraulic & Pneumatic Lines, Fittings and Hoses	1.00	2.50	4.00
Propellant Supply and Vent Components	1.00	1.50	4.00

Table 3.1-8. Minimum Margins of Safety

	Minimum Margin of Safety
Fasteners in shear	+ .15
Bolts in tension	+ .50
Fittings	+ .15
Lugs	+ .25
Welds - Electron Beam	+ .15
Welds - Other	+ .50 (Dependent on inspection procedure)
Bonded Joints	+ .50

Margins of safety less than 2.0 shall be indicated numerically. Those greater than 2.0 may be listed as "high".

For purposes of determining design margin of safety, material properties shall be obtained from MIL-HDBK-5 and MIL-HDBK-23A. Other data sources may be used when specifically approved by the General Electric Company. For purposes of analysis design margins of safety shall be based on minimum thicknesses and on the "A-values" of MIL-HDBK-5 for single load path structures where compressive buckling is the failure mode. For all other conditions the minimum guaranteed properties and the nominal thickness shall be used.

3.2.1.2.4 Alignment

Mounting structures shall be capable of maintaining the component alignment requirements within the error budget allocations shown in Table 3.2-9, throughout orbital life after exposure to the environments of paragraph 3.2.5.

Mechanical alignment includes all factors resulting from load deflections during orbit maneuver conditions, permanent set hystereses as a result of launch ascent, as well as mechanical adjustment uncertainties and manufacturing assembly tolerances.

Thermal alignment shall include distortion of mounts and supporting structure due to the surrounding environment, overall distortion of the structure and thermal load creep.

Table 3.2-9. Alignment Requirements

Item	Aligned to	Required Alignment Arc-Minutes/Axis
Attitude Control S/S (ACS)	Vehicle Ref.	TBD
Momentum Wheels	ACS Ref.	30
Inertial Reference Unit (IRU)	ACS Ref.	30
Star Sensor Unit	ACS Ref.	TBD
Magnetic Torquers	ACS Ref.	30
Sun Sensor	Vehicle Ref.	30
Secondary Propulsion S/S	Vehicle C.G.	TBD
Instrument Section (IS)	Vehicle Ref.	TBD
Wide Band Antenna	I.S. Ref.	TBD
Thematic Mapper (TM)	I.S. Ref.	TBD
Multi-Spectral Sensor (MSS)	I.S. Ref.	TBD

3.2.1.3 Mechanism Functions

3.2.1.3.1 Module Latch Mechanisms

Module latch mechanisms shall be located at each of the four corners of the subsystem section structure and the module frame assemblies to firmly attach and/or lock the module at the spacecraft interface. These mechanisms shall also interface with the Module Exchange Mechanism (MEMS) of the Space Shuttle Flight Support System.

3.2.1.3.2 Launch Vehicle Separation Device

The Launch Vehicle Separation Device shall retain the spacecraft to the launch vehicle adapter during powered flight, release the circumferential retention band, and apply an ejection force to the spacecraft to separate it from the launch vehicle within the velocity requirements specified in Document TBD.

3.2.1.3.3 Solar Array Retention/Deployment/Retraction

A. Solar Array Retention. The solar arrays shall be retained in the stowed position as shown in Drawing TBD. The retention assembly shall provide a means to isolate the panels from the spacecraft structure so as to limit vibration response levels of the panels in the resonance range of TBD to TBD Hz.

B. Release, Deployment, and Retraction.

   a. Contamination. The release mechanism shall generate no loose mechanical debris nor gaseous contaminants.

   b. Deployment/Retraction Time. The solar array shall be fully deployed (or retracted) within TBD seconds after receipt of power from the EPS.

   c. Deployment/Retraction Forces. The solar array deployment/retraction mechanism assembly shall limit the amount of energy to be absorbed when deployed (or retracted) to a maximum of TBD inch-pounds at the hinges. A positive margin of deployment torque shall be provided.

   d. Position and Clearances. The solar array panels shall be capable of being deployed from their stowed position (or retracted from their fully extended position) with no possibility of striking the spacecraft or its appendages.

e. Stops, Latches, and Releases. Stops, latches, and releases shall be provided at each axis of rotation to limit rotation, assure a tight joint, and permit release prior to retraction.

3.2.1.3.4 Solar Array Drive and Power Transfer

The structure subsystem shall provide the mechanical and electro-mechanical devices to drive the deployed solar array panels for acquiring and tracking the sun and to transfer raw electrical power and electrical signals across the rotary joint. These devices when assembled shall be referred to as the Solar Array Drive and Power Transfer Assembly (SADAPTA). Detail requirements, conforming to the requirements identified herein, shall be as presented in Table 3.2-10.

3.2.1.3.5 Wideband Antennas

The Wideband Antennas shall be retained in a stowed position during powered flight, shall be deployed to an earth-facing position in orbit, and shall be capable of being steered approximately $\pm 60^{o}$ about two axes.

3.2.1.3.6 Tracking and Data Relay Antenna (TDRSS)

The TDRSS antenna shall be retained in a stowed position during powered flight, deployed and unfurled to 8 feet diameter in orbit, pointed at a synchronous tracking and data relay satellite, and refurled and retracted for Space Shuttle retrieval.

3.2.1.4 Useful Life

The useful life of this subsystem shall be a minimum of TBD years starting with acceptance of the spacecraft by the procuring agency in accordance with the following operational phases.

Table 3.2-10. Solar Array Drive Requirements

Rotational Capability	Continuous rotation One direction Negative vehicle pitch
Position Accuracy	$\pm 5^{\circ}$ to sun line
Angular Rate	4°/minute
Slew Rate	40°/minute
Position Feedback	Output shaft position indication @ 1200 pulses/rev.
Power Transmission	TBD power circuits TBD signal circuits
Size	TBD
Weight	TBD
Life	2 years

Storage:	TBD years
Pre-Launch:	TBD years
Launch:	Negligible
Orbit:	2 Years

The useful life of individual components within the Structural Subsystem shall include additional time accrued prior to incorporation into the subsystem (i.e., transportation, handling, storage, and testing at the component level).

3.2.2 PHYSICAL CHARACTERISTICS

3.2.2.1 Configuration

The overall configuration of the EOS, packaged within the shroud constraints of the Delta launch vehicle, shall be similar to the arrangement shown in Figure 3.2-1.

The EOS Structure Subsystem shall be comprised of a set of structural modules and electro-mechanical mechanisms some of which shall form the GPSS; the remainder form the MPSS.

3.2.2.2 GPSS

The General Purpose Spacecraft Segment shall be the basic portion EOS - essentially identical for all vehicles, regardless of the mission. It shall be a single integrated assembly of structural modules and mechanisms shown in Figure 3.2-2.

3.2.2.2.1 Subsystem Section Structure

The Subsystem Section Structure of the GPSS, Figure 3.2-3, shall interface with the Secondary Propulsion Module Structure, and shall support the solar array support structure, the transition frame structure, and the three subsystem module frame assemblies. The overall dimensions shall be constrained by the 86" diameter Delta shroud envelope and the volume and shape characteristics of the MPSS and subsystem components. The general arrangement of the Subsystem Section Structure with respect to the other subsystems and shroud constraints shall be as shown in Figure 3.2-4.

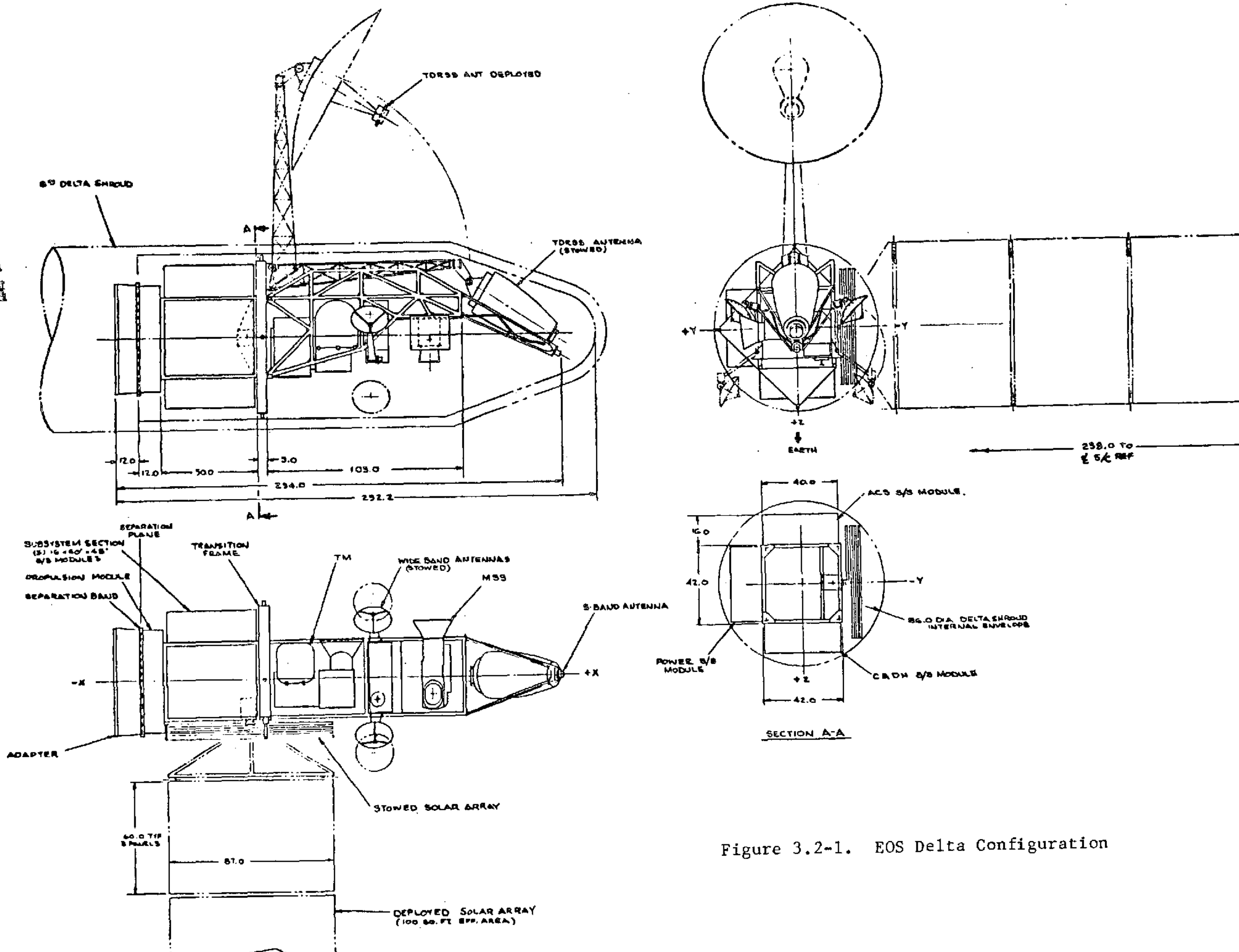


Figure 3.2-1. EOS Delta Configuration

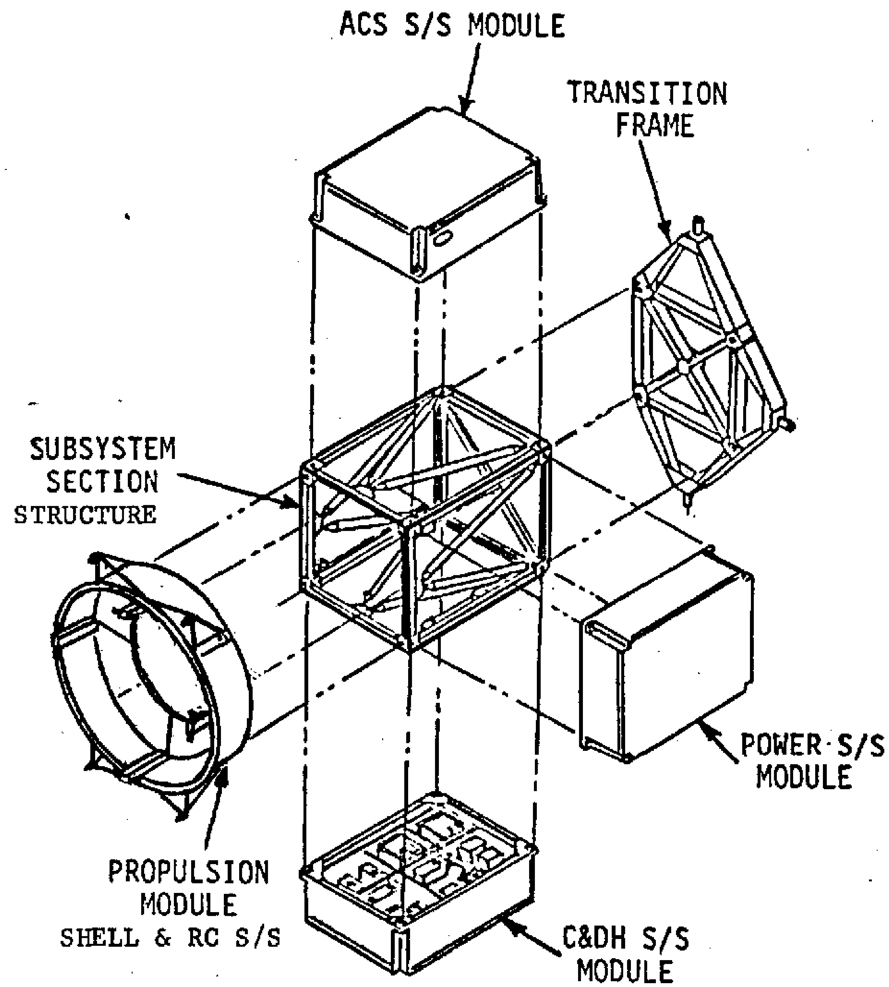


FIGURE 3.2-2. GPSS CONFIGURATION

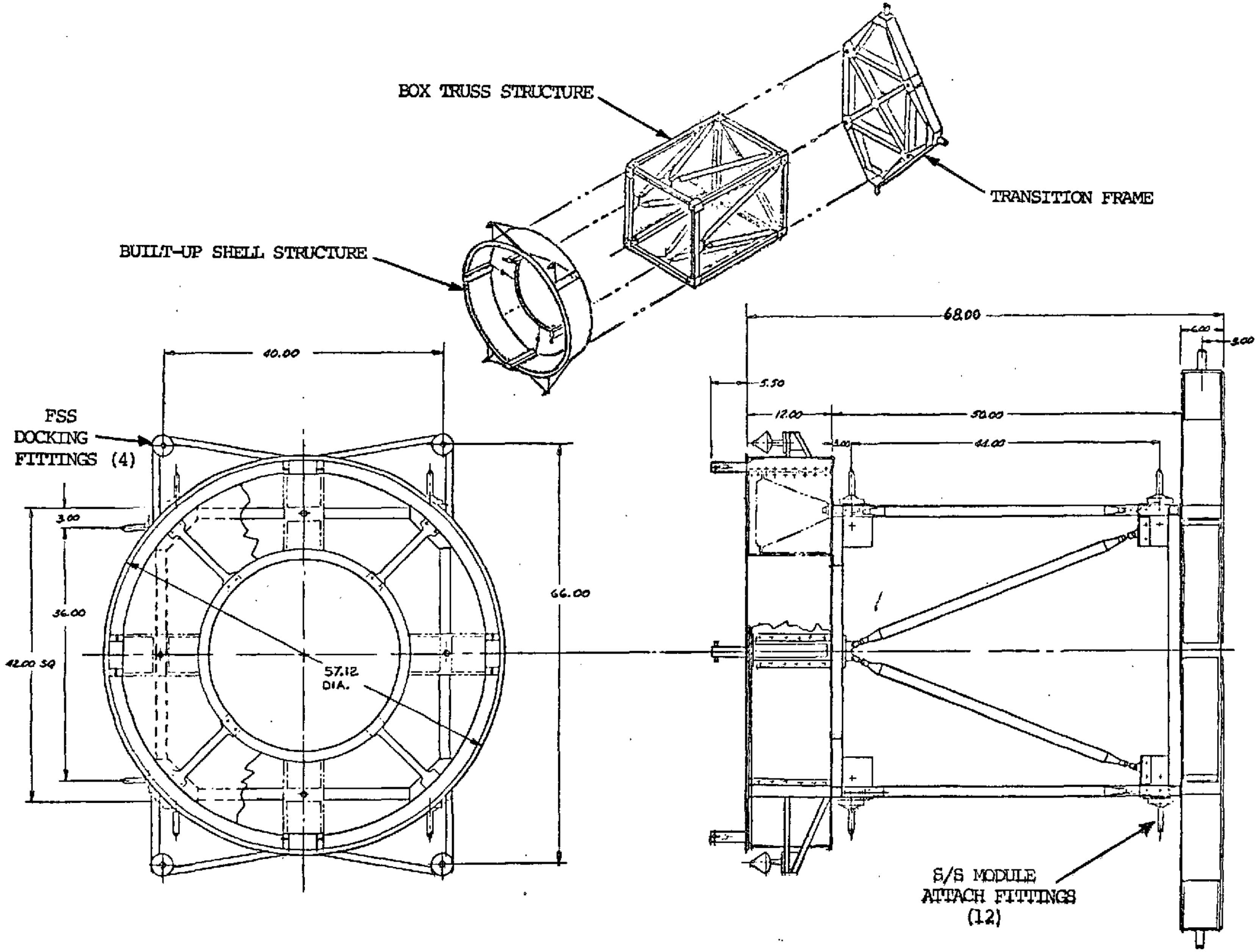


Figure 3.2-3. Subsystem Section Structure

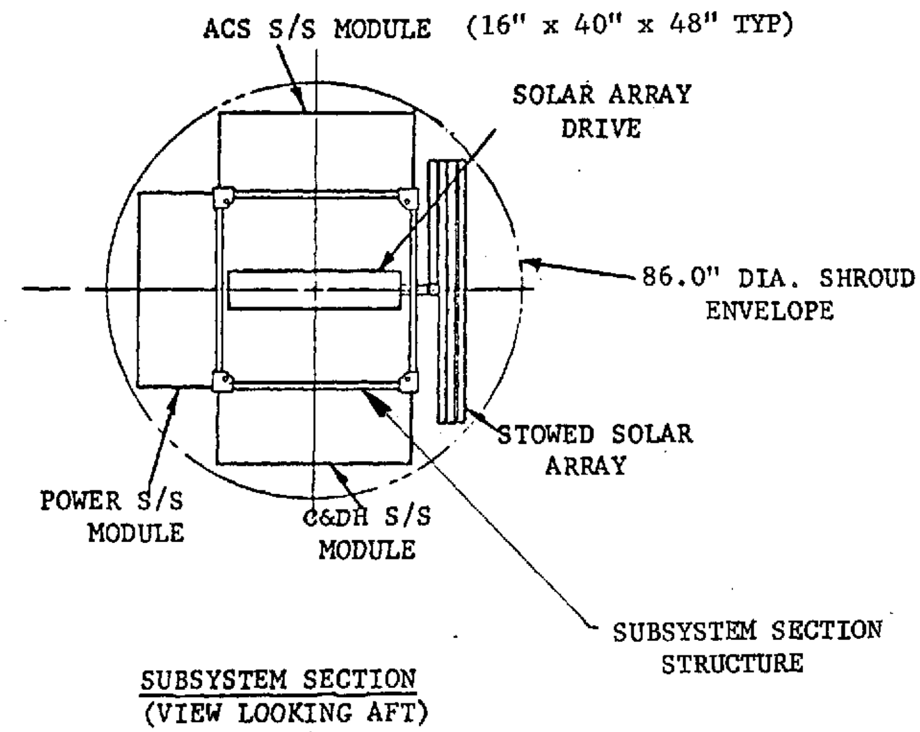


Figure 3.2-4. Structure Subsystem/Delta Shroud Envelope Constraint

3.2.2.2.2 Propulsion Module Structure

The cylindrical shaped Propulsion Module Structure shall support all the GPSS and MPSS designated SPS components as an integral module assembly. The structural configuration shall accommodate a range of hydrazine propellant tank sizes and thruster locations. Figure 3.2-5 illustrates the Propulsion Module Structure for the Reference EOS. The structure shall interface via Vee-band separation device with the launch vehicle adapter, and with a four-point attachment to the Subsystem Structure described in 3.2.2.2.1. The module, with SPS components installed, shall be removable from the spacecraft to permit assembly, checkout, and test of the SPS.

3.2.2.2.3 Subsystem Module Substructure

The general arrangement of the electronic and mechanical components installed in the Subsystem Module Substructure shall be in accordance with Figure 3.2-6. The 48" x 40" x 16" module size shall be used for the ACS, Power, and C&DH subsystems. Components shall be mounted to the inner face of the aluminum honeycomb outer panel which shall be integrally stiffened as required to support individual components. Provisions shall be made for installation of harnesses and insulation blankets.

3.2.2.2.4 Transition Frame

A structural transition frame shall be attached to the forward corners of the Subsystem Section Structure, support the Instrument Section of the MPSS, and provide for a three-point retention interface with the Space Shuttle for retrieval. The Transition Frame geometry, support conditions, and general arrangement with the Shuttle Bay Fuselage Frame shall be as shown in Figures 3.2-7, -8, and -9.

3.2.2.2.5 Module Latching Mechanism

The Module Latching Mechanism shall be similar to the GSFC design shown in Figure 3.2-10. The latch body shall be attached to the four inboard corners of the module frame assemblies and shall engage mating screws fixed to the Subsystem Section Structure. The latching mechanism shall be a threaded nut which shall pull and preload the module against the spacecraft mounted screw. The mechanism shall be capable of being operated by the Shuttle MEM installation mechanism.

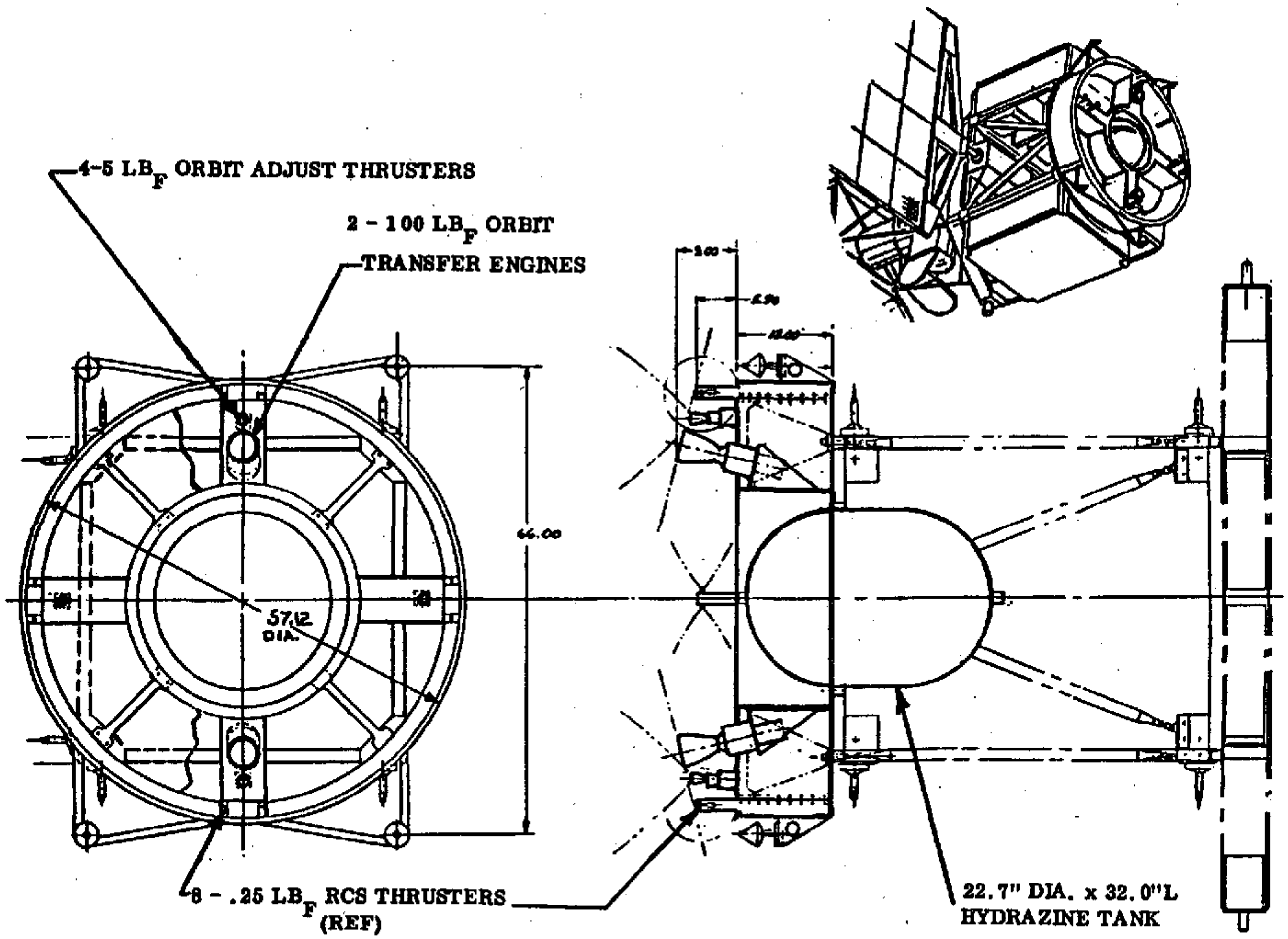


Figure 3.2-5. Propulsion Module Structure

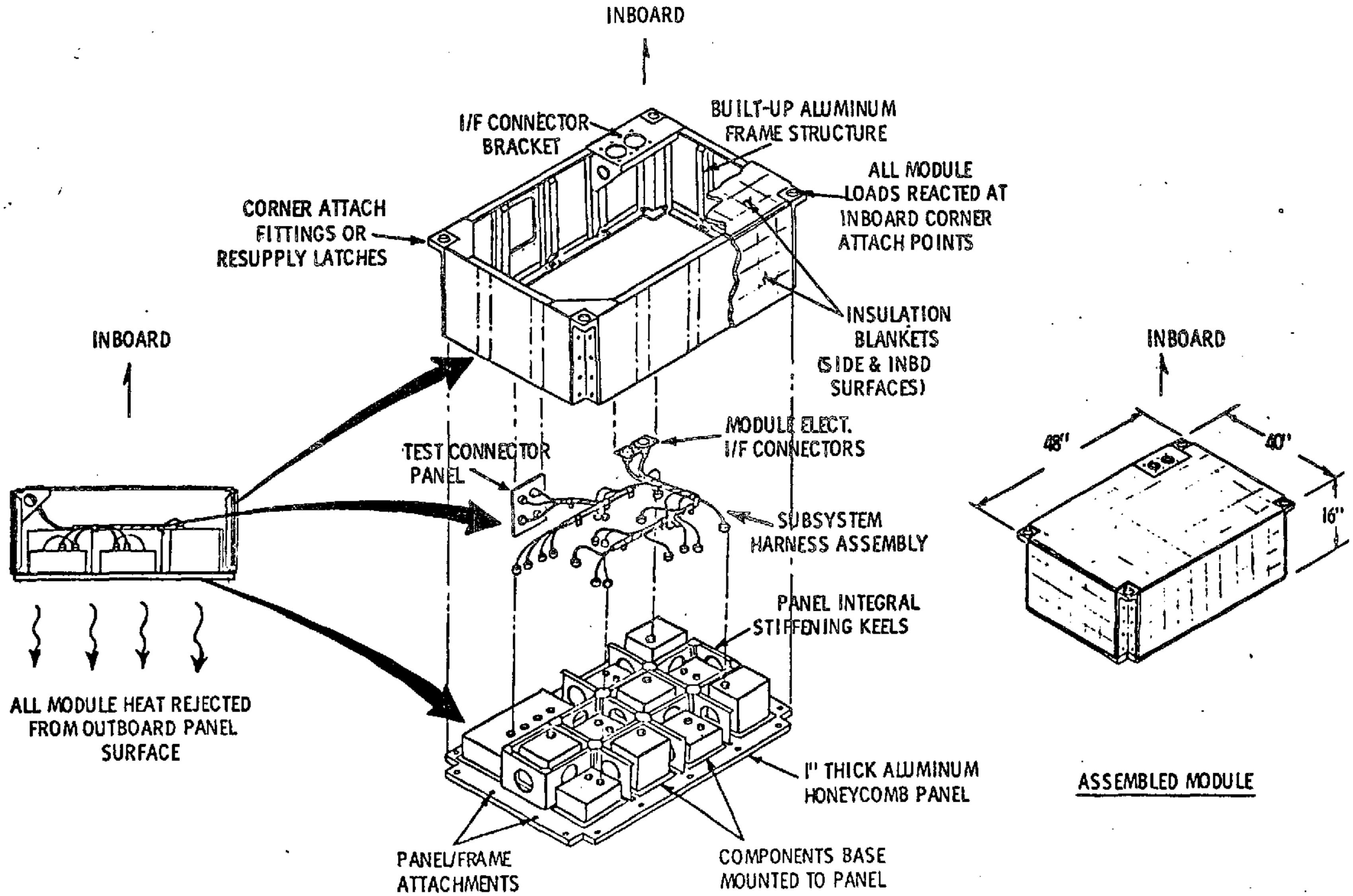


Figure 3.2-6. Subsystem Module Structure

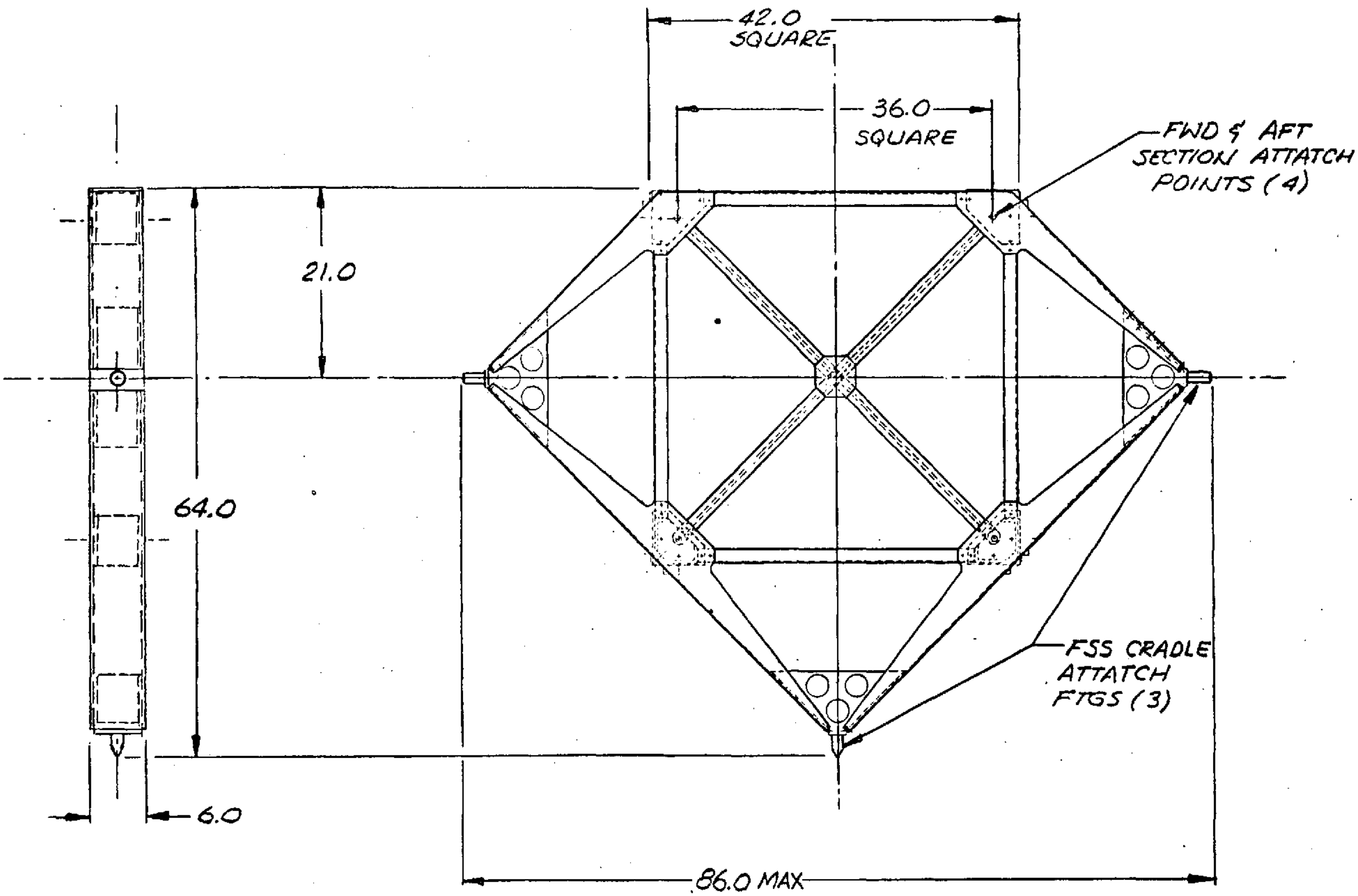


Figure 3.2-7. Transition Frame Geometry

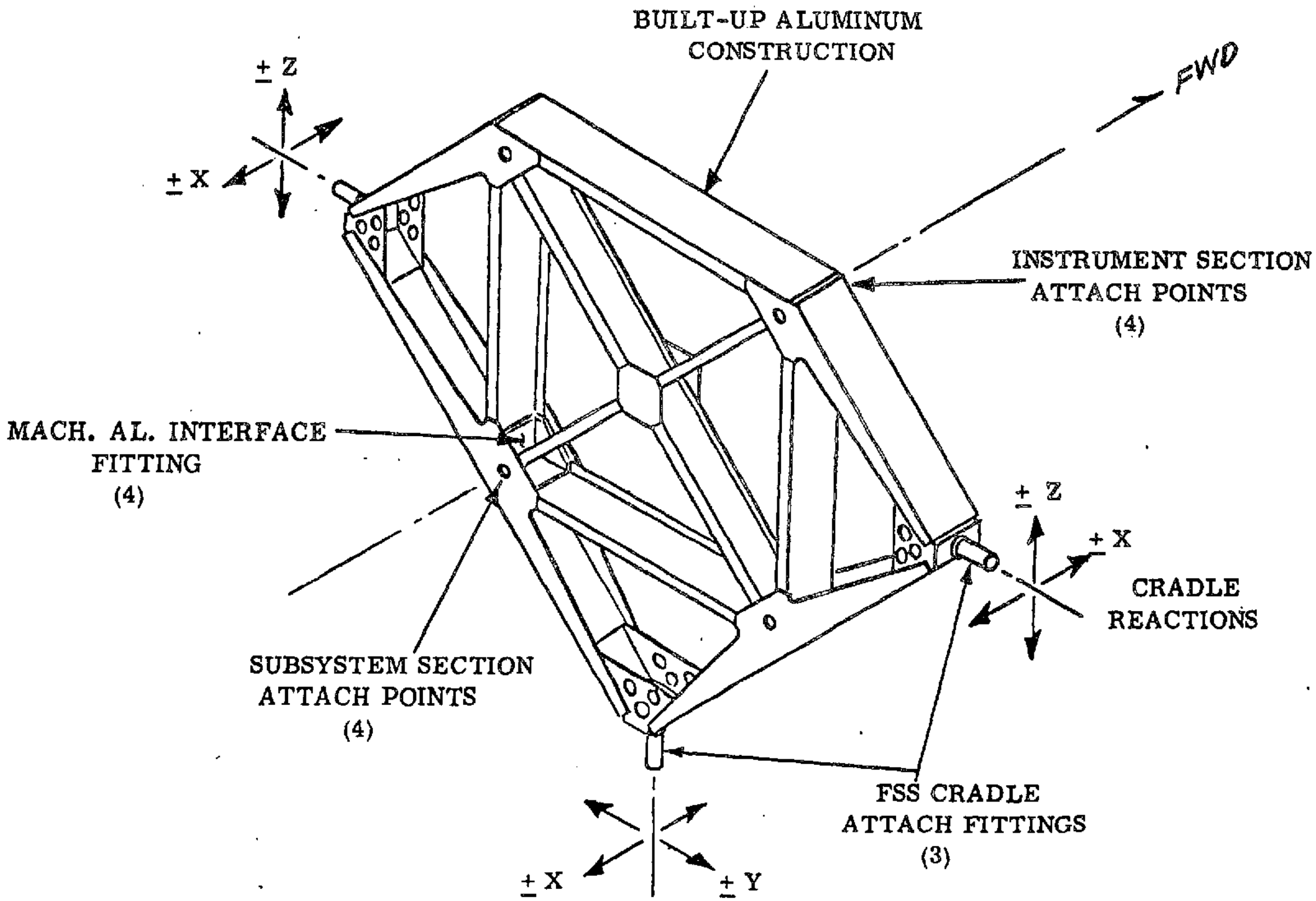


Figure 3.2-8. Transition Frame Loading Condition

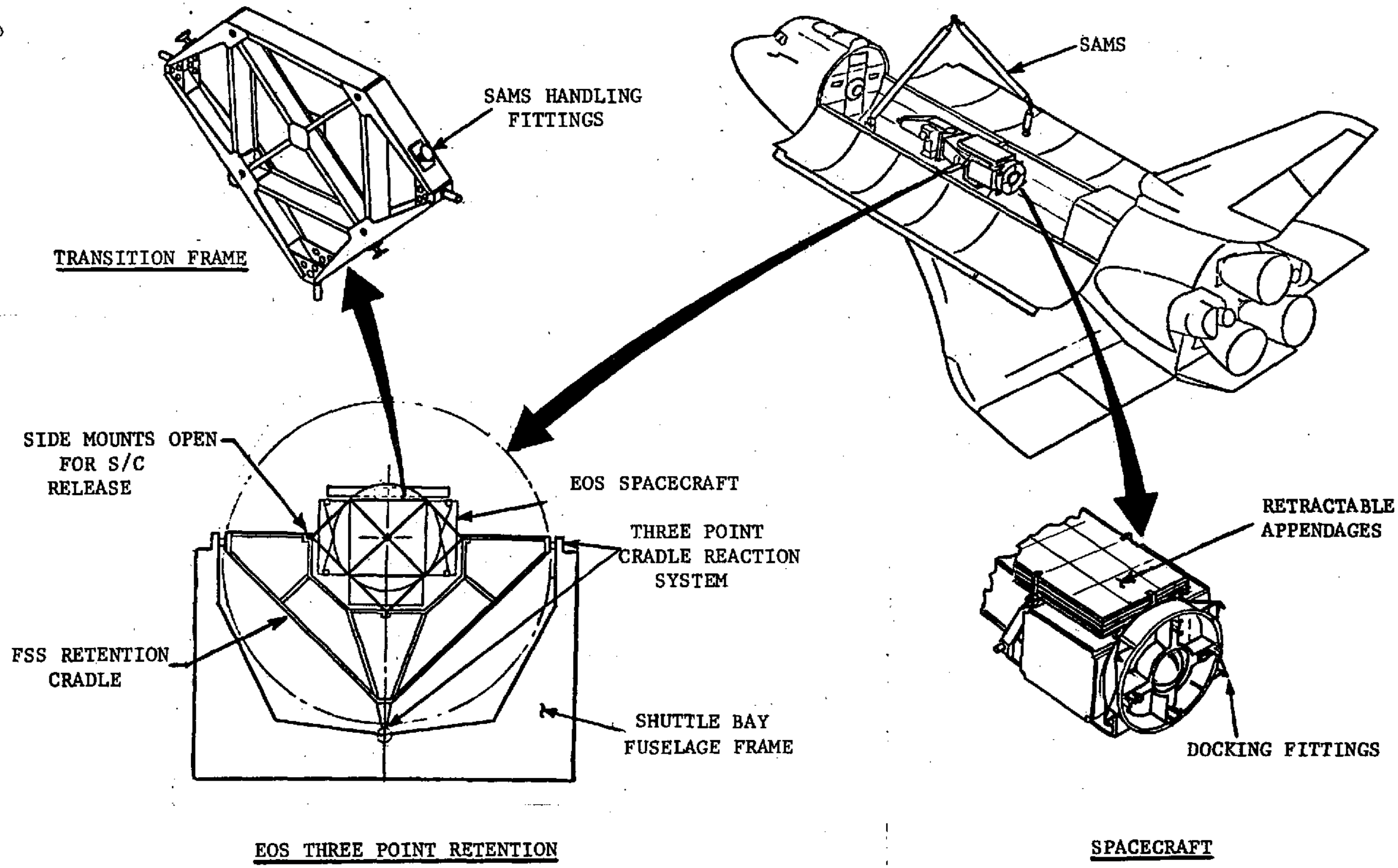


Figure 3.2-9. EOS/Shuttle General Arrangement

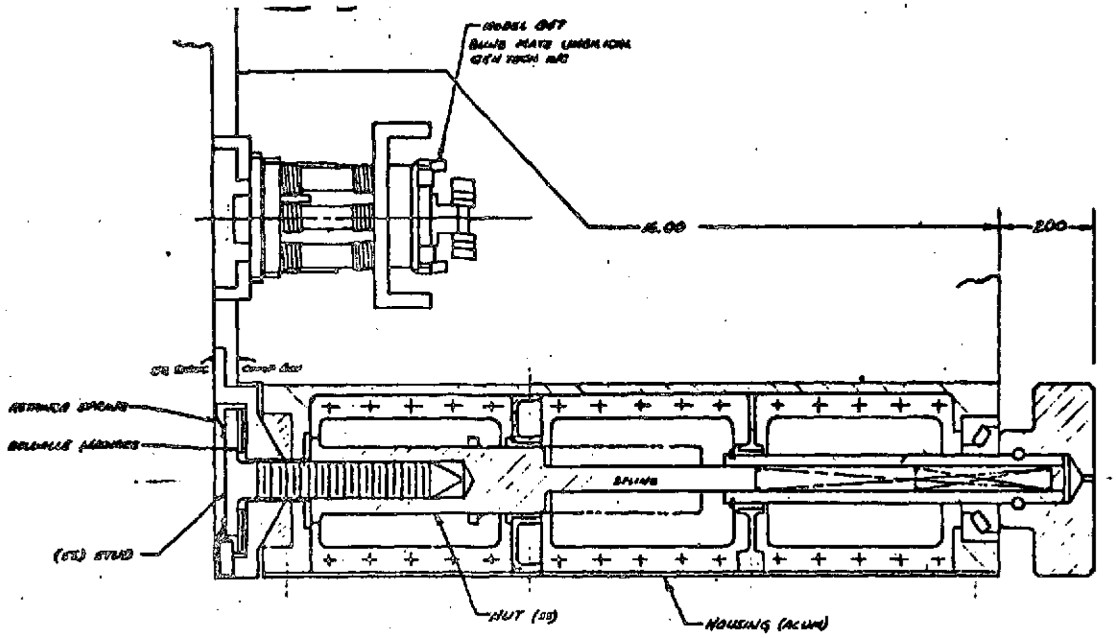


Figure 3.2-10. Module Latching Mechanism

3.2.2.3 MPSS

The Mission Peculiar Spacecraft Segment shall be that portion of the EOS which must be uniquely configured to comply with specific mission requirements. It shall be a set of integrated assemblies consisting of semi-standardized structural modules and mechanisms as shown in Figure 3.2-11. The major sub-segments of the MPSS Structure System are:

- o Launch Vehicle Adapter
- o Solar Array Assembly and Drive Mechanisms
- o Instrument Section Structure
- o Support for the MPSS SPS components

Selection of structural elements, mechanisms, thrusters, solar array panels, etc. shall utilize as far as possible commonality of components and subassemblies.

3.2.2.3.1 Launch Vehicle Adapter

The Launch Vehicle Adapter shall be designed to interface with the Delta 2910 (or Titan III B/NUS) booster, support the EOS during launch, and interface with the Vee-band separation device described in 3.2.2.3.5.

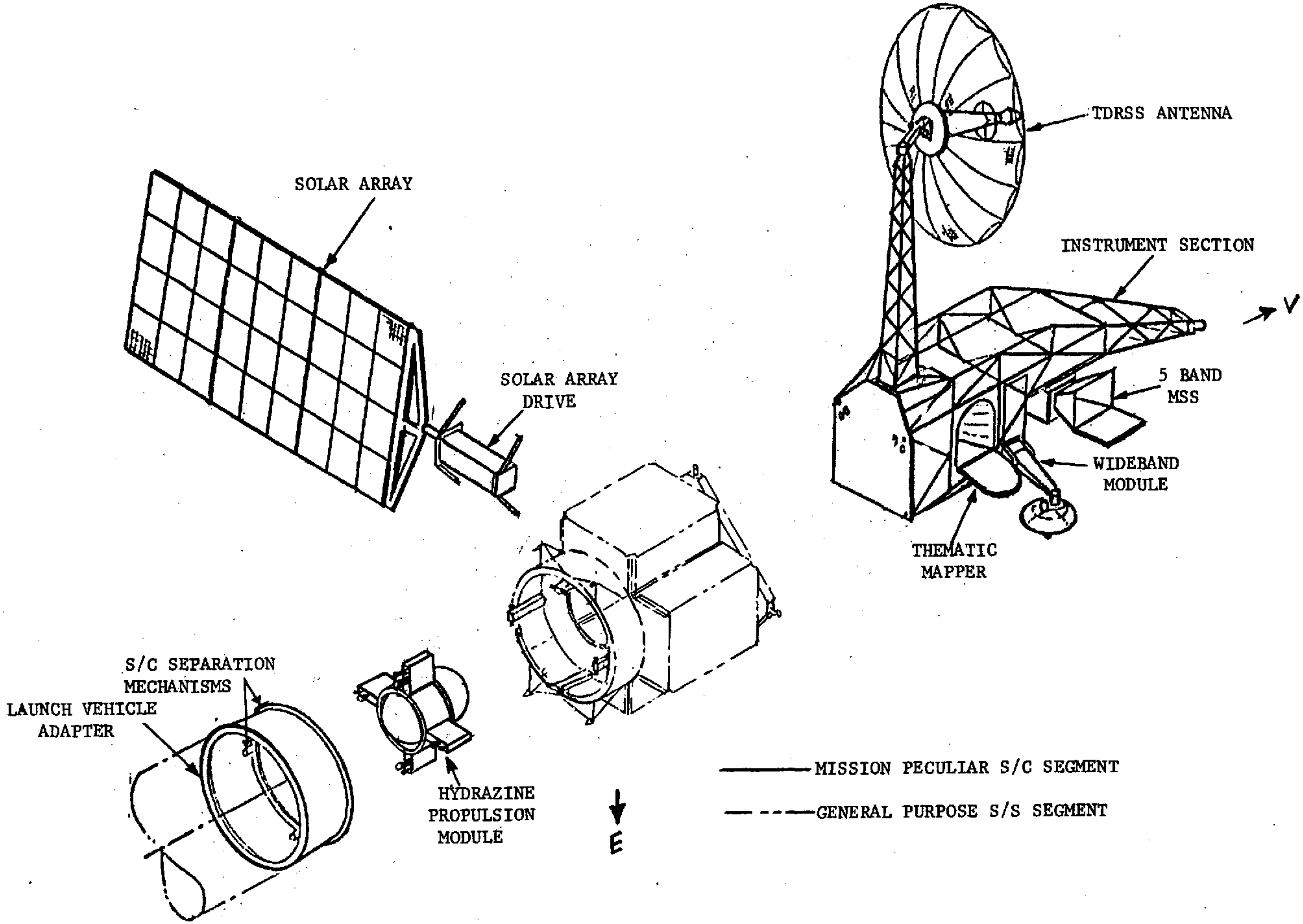


FIGURE 3.2-11. MISSION PECULIAR SPACECRAFT SEGMENT

3.2.2.3.2 Solar Array Assembly and Drive Mechanisms

The Solar Array Assembly (comprised of the yoke, solar array deployment mechanism, and solar array panel assemblies) will be capable of being installed and removed from the spacecraft as a separate and integral panel stack. Ground testing of solar array deployment and vibration testing of the deployed solar array may use assist devices, counterbalances and/or special support devices to counteract the 1 g environment. The arrangement for storing, deploying/retracting, and for orienting the fully deployed array shall be as shown in Figure 3.2-12.

The solar array size shall be mission dependent having differing areas, form factors, and orientation requirements. Standardized aluminum honeycomb substrate panels shall be mounted to a built-up frame structure illustrated in 3.2-13.

3.2.2.3.3 Instrument Section Structure

The EOS Instrument Section Structure mounted to the forward face of the Transition Frame of the GPSS shall be a mission unique structure configured to support the specific mission payloads. The structural arrangement and construction of these Sections for alternate payloads shall be similar to the structure designed to support the Thematic Mapper and MSS instrument and TDRSS shown in Figures 3.2-14 and 3.2-15.

The overall dimensions of the Instrument Section Structure with the instruments installed shall fit within an 86-inch diameter shroud envelope (for the Delta launch) as shown in Figures 3.2-16 and 3.2-17.

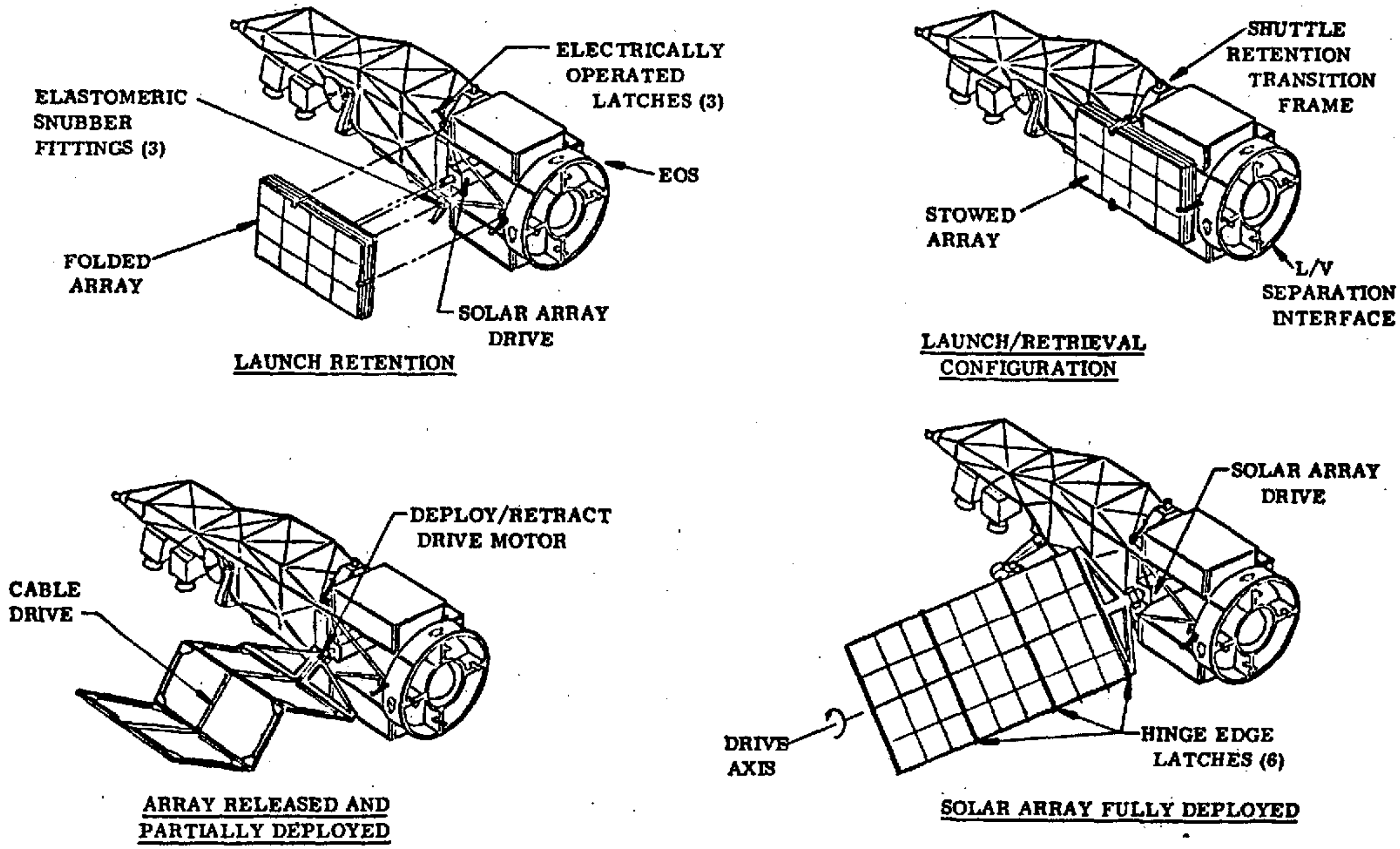


Figure 3.2-12. Solar Array Retention/Deployment/Retraction

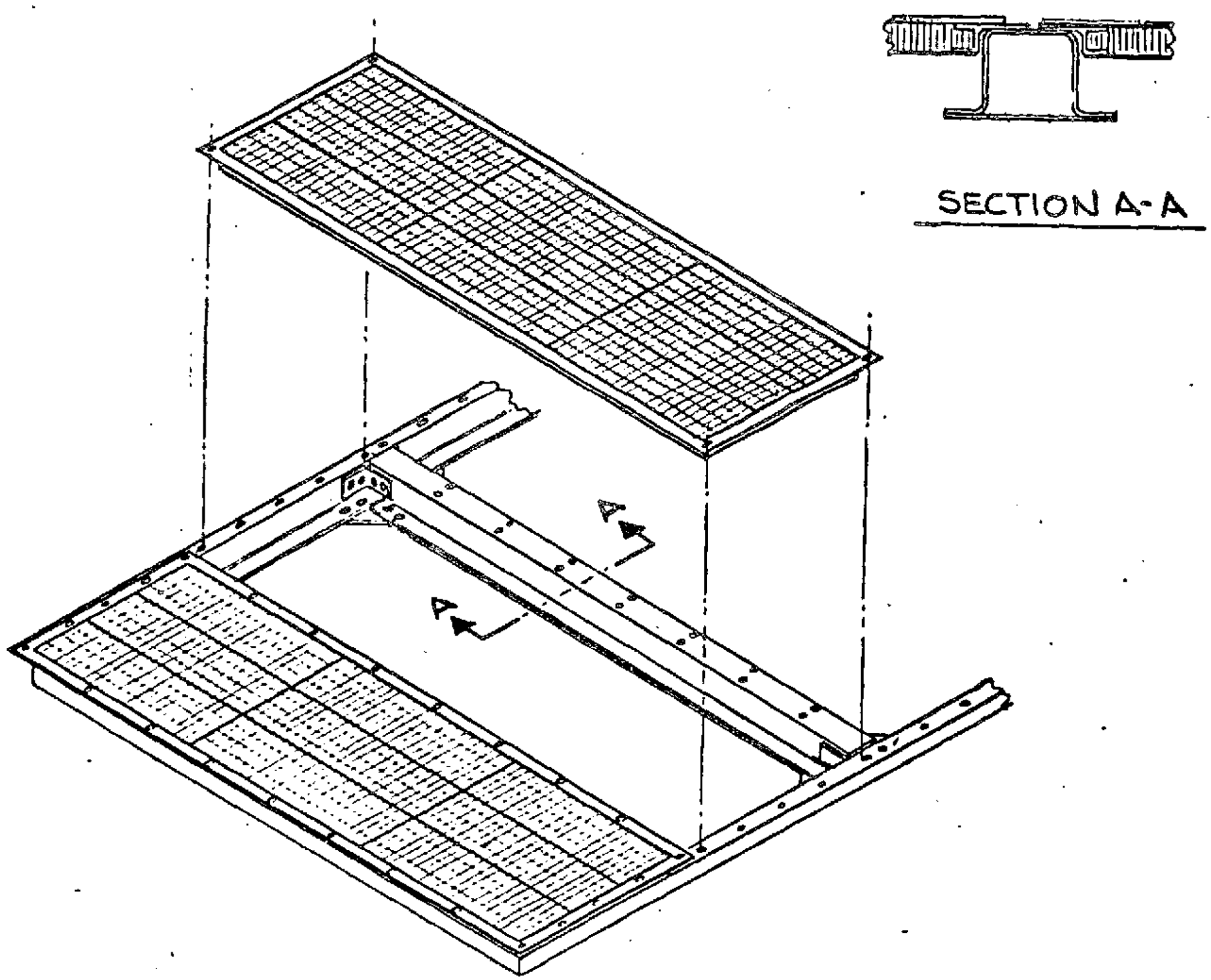


Figure 3.2-13. Solar Array Panel

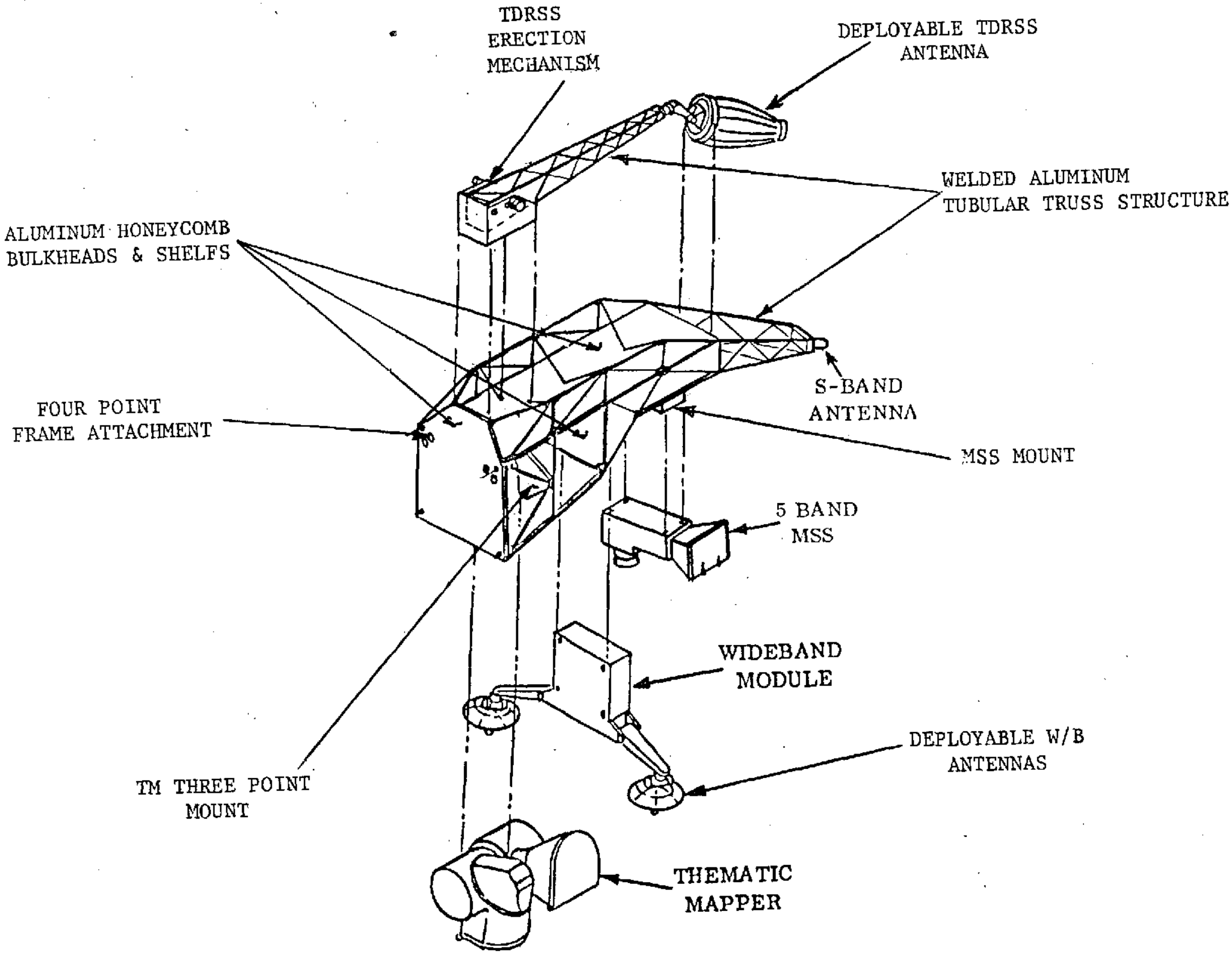


Figure 3.2-14. Reference Instrument Section Structural Arrangement

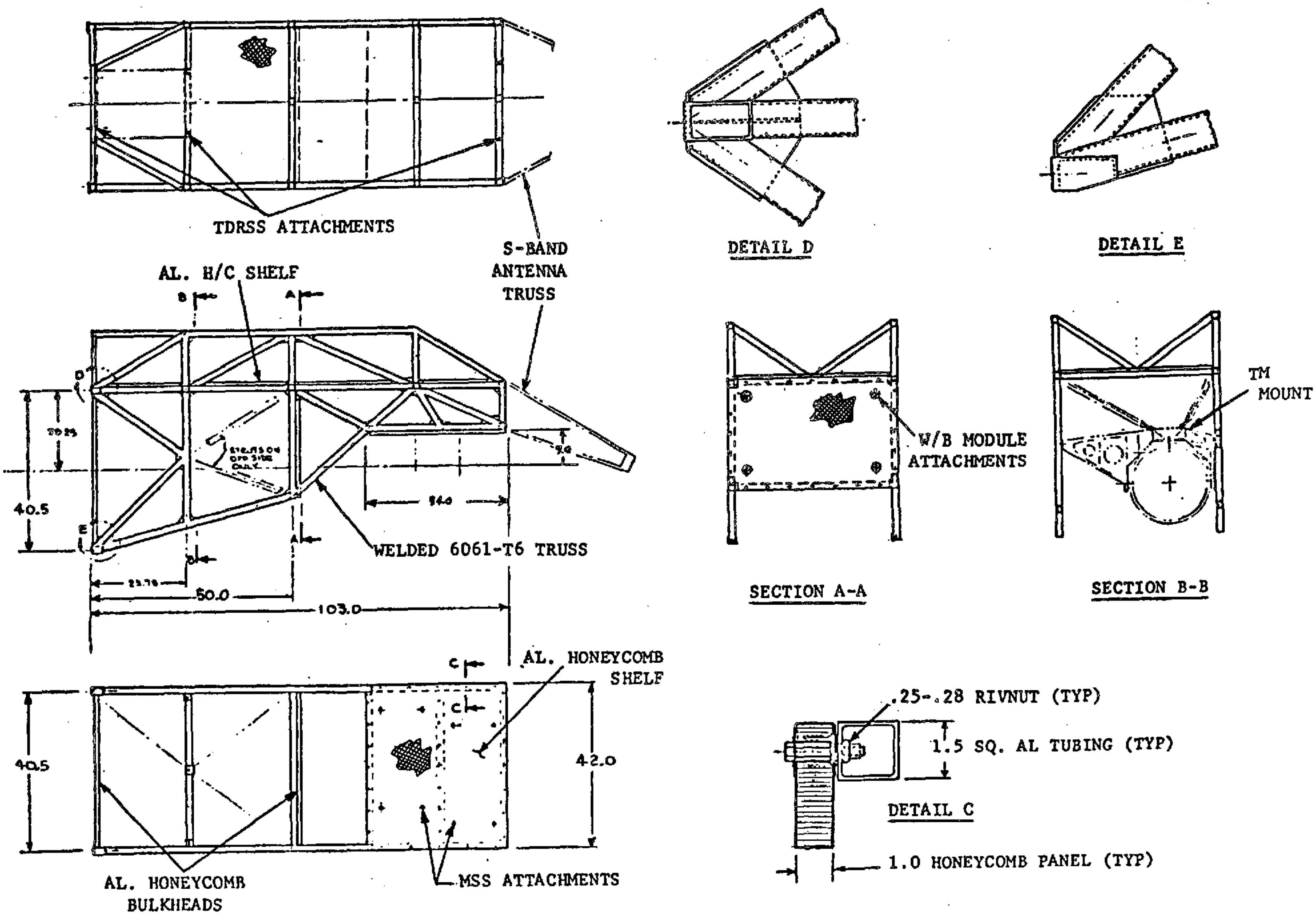


Figure 3.2-15. Instrument Section Structure

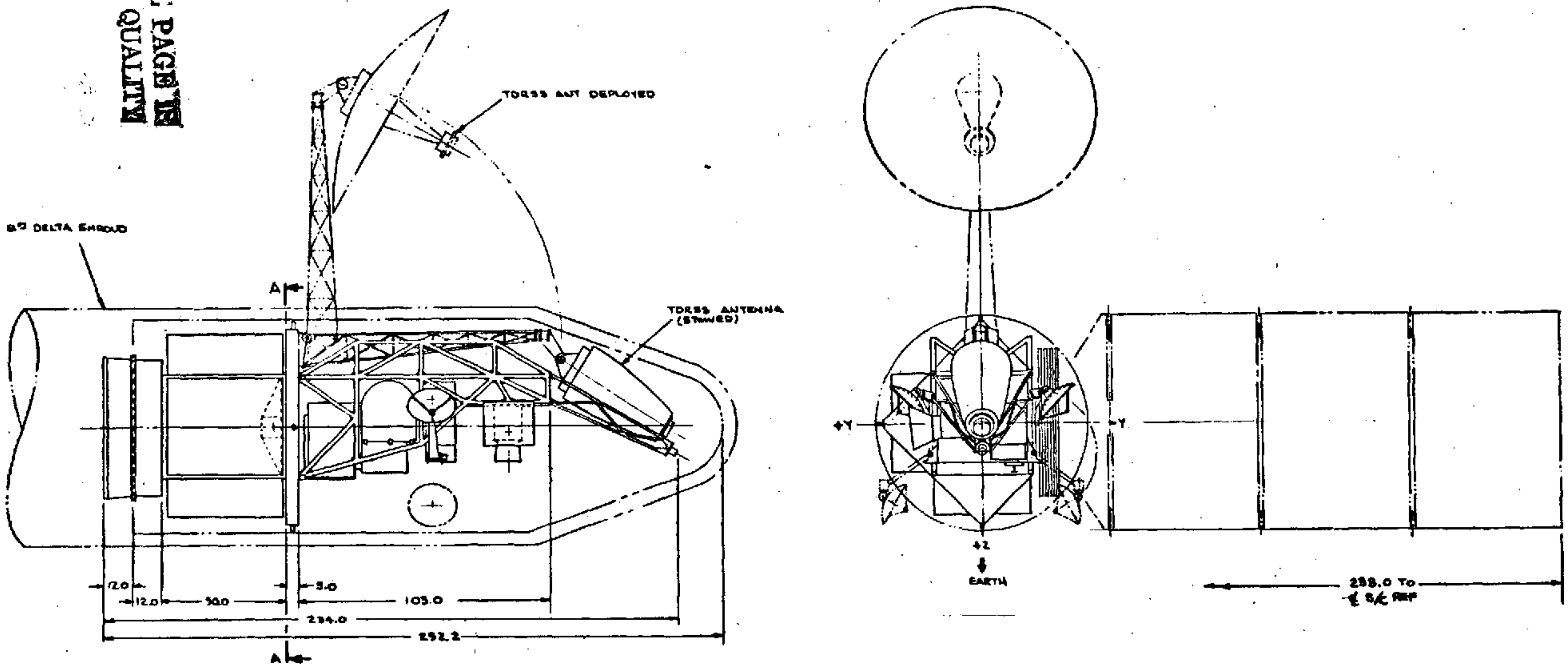


Figure 3.2-16. Instrument Section/Delta Shroud Constraints

3.2.2.3.4 Launch Vehicle Separation

A. Vee-Band. The launch vehicle adapter/propulsion module substructure interface shall accommodate a Vee-band circumferential ring shown in Figure 3.2-17. The joint shall be pre-loaded to prevent gapping under load and shall be separated by pyro activated redundant bolt cutters on the band. The separated band halves shall be retained on the launch vehicle adapter after separation to eliminate space debris.

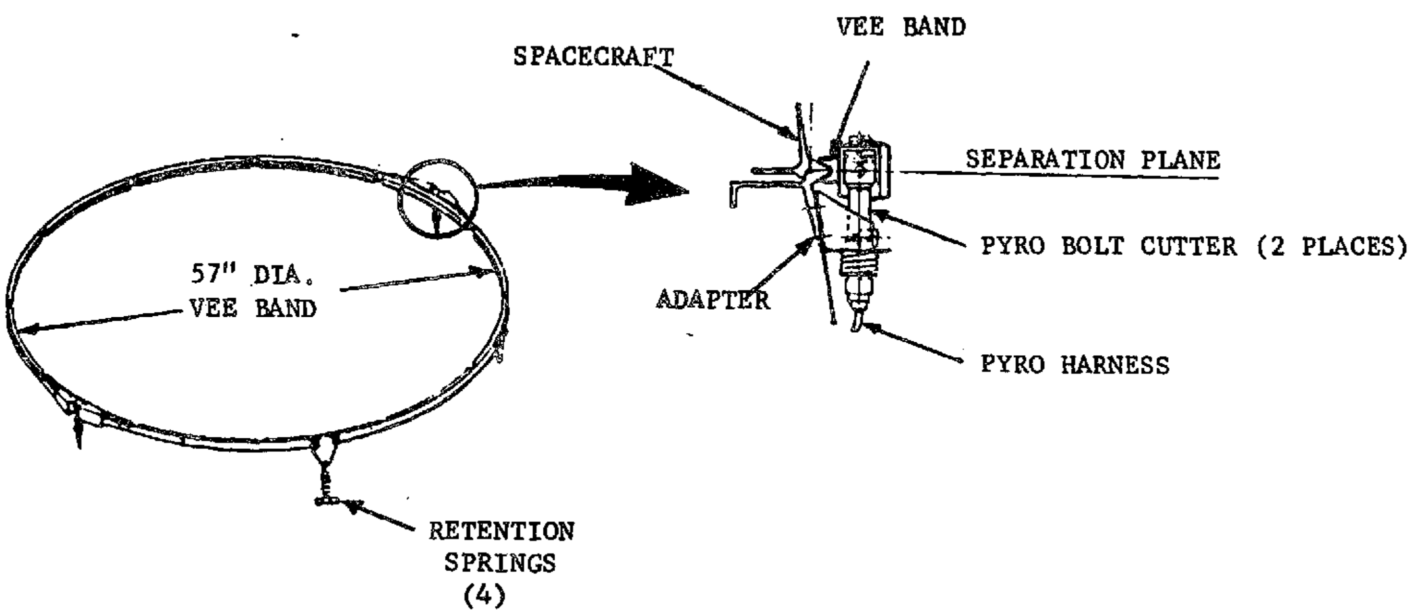


Figure 3.2-17. Vee-Band Retention/Separation

B. Separation Springs. Separation velocity shall be achieved by the use of Spring Assemblies that shall be adjustable so that center-of-mass offsets in the spacecraft can be compensated by spring forces to insure minimum tip-off rates.

3.2.2.3.5 Solar Array Deployment/Retraction

A. Synchronized Motion. The solar array panels shall be deployed (or retracted) by an arrangement similar to Figure 3.2-18. A motor drive mechanism shall provide a control on both extend and retract rates such that both motions shall be performed smoothly at a constant rate within a period of approximately TBD minutes.

B. Latching/Unlatching. Each hinge line shall be provided with a zero "slop" and backlash latching device on each side of the panel which shall automatically securely lock the panels against fixed stops at the end of the travel. These latches shall be capable of being set into an unlatched condition on command with a DC voltage input of 28 VDC of 2 seconds maximum pulse width.

3.2.2.3.6 Solar Array Drive

The solar array drive shall provide the rotation of the solar panels about the pitch axis as required to track the sun.

Referring to Figure 3.2-19, the drive for a reference EOS shall consist of a 1.8° stepper motor, a 100 to 1 harmonic drive speed reducer followed by a further gear reduction of approximately 6:1. The output shaft shall be hollow and concentric with the main paddle shaft being connected to it by means of a wrap spring overriding clutch. There shall be two identical drives (one redundant coupled to the paddle shaft). Hence, if a failure occurs in one drive, the second drive shall be energized and shall inherently transmit torque to the paddle shaft through its overriding clutch. Each drive shall have an output torque rating of TBD ft-lbs and shall reliably accommodate an inertia load of TBD slug-ft at 100 pulses per second to the stepper motor ($\cong$ 3 rpm at the output).

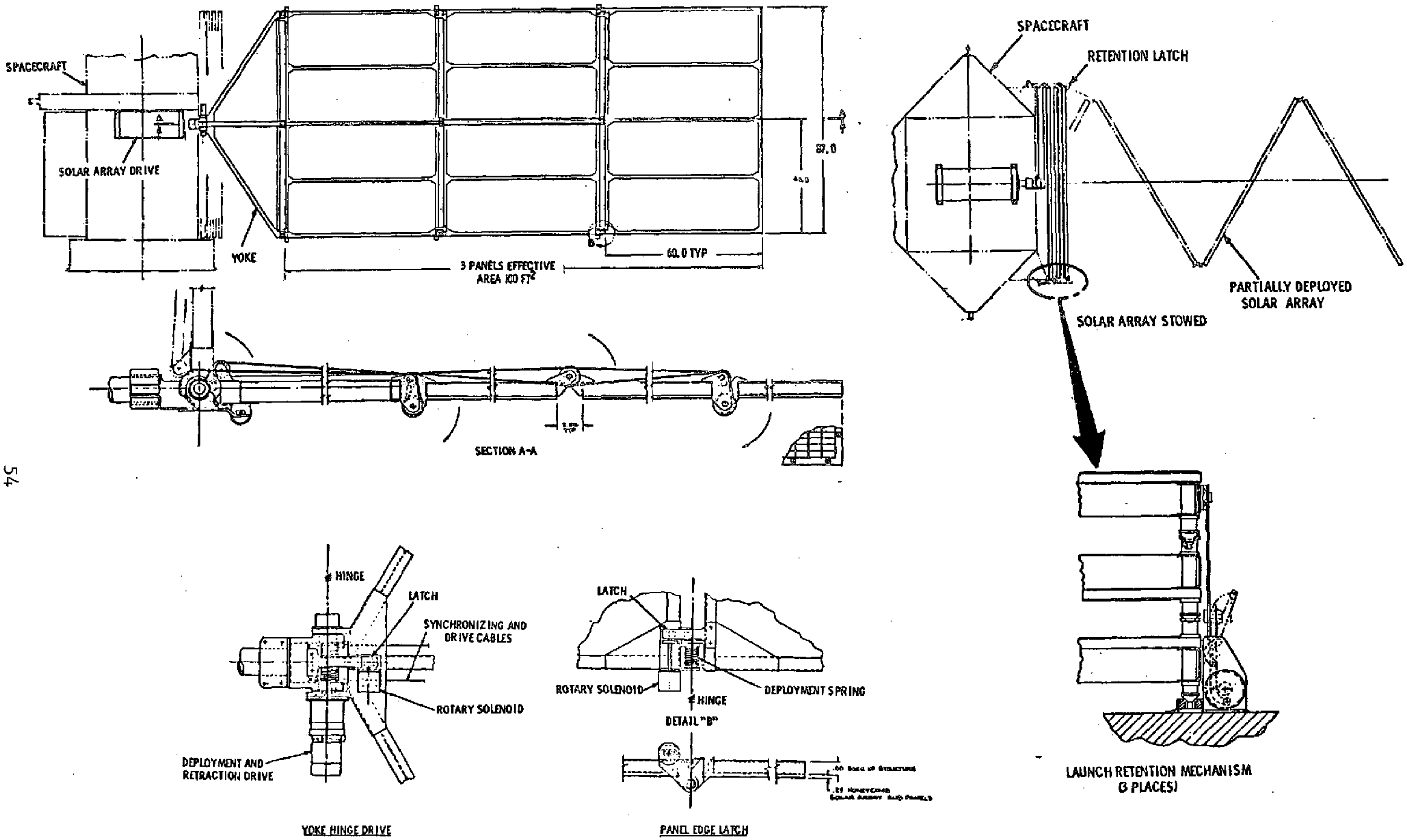


Figure 3.2-18. Solar Array Deployment/Retraction Mechanism

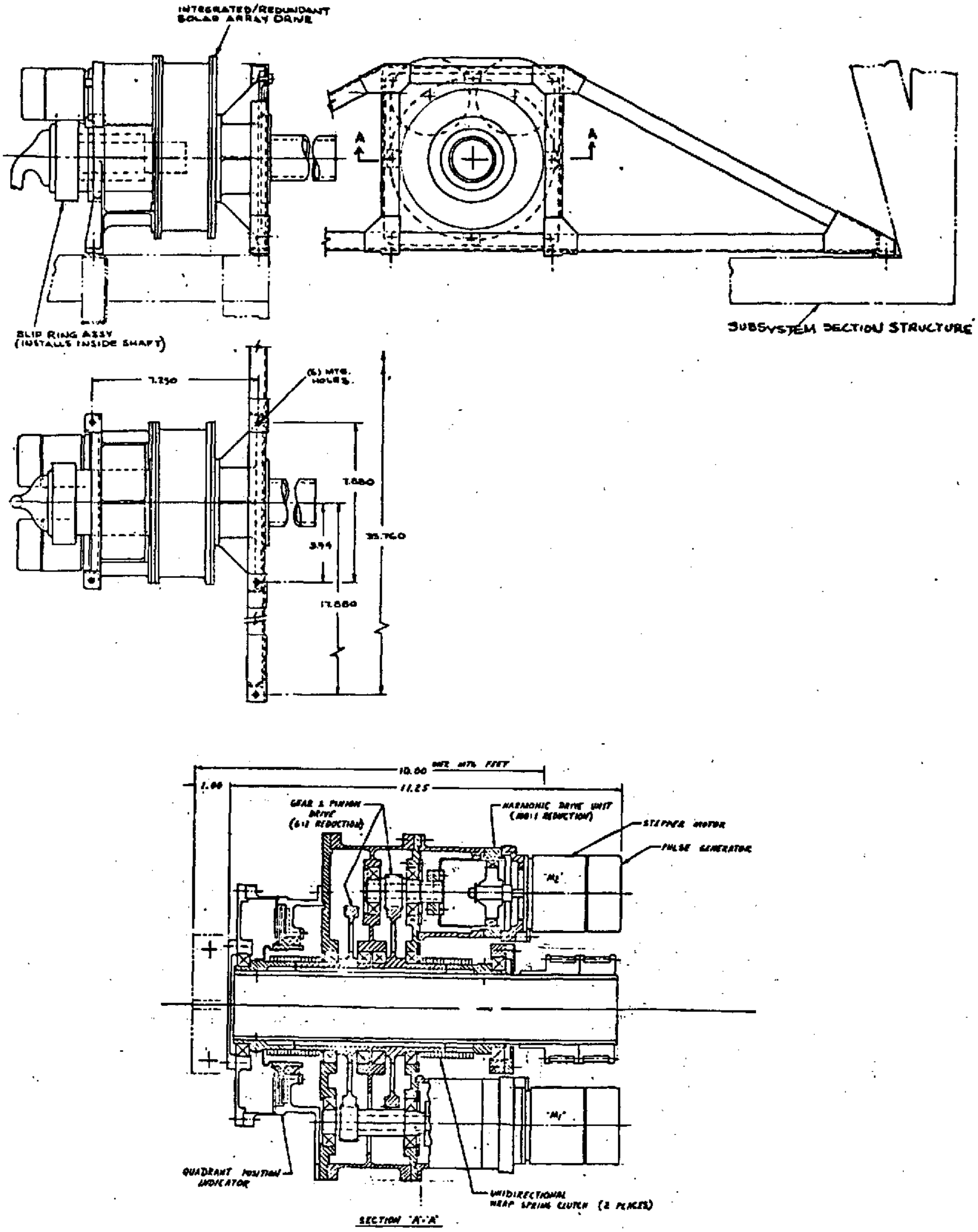


Figure 3.2-19. Solar Array Drive

3.2.2.3.7 Wideband Antenna

A. Configuration. Support for the wideband antenna/gimbal mechanisms and the drive electronics shall be integrated into a single module comprising a portion of the instrument section of the spacecraft. The drive subsystem, which shall orient the antenna about two orthogonal axes $\pm$ TBD$^{\circ}$, shall be identical for each of the two wideband antennas.

B. Gimbal Rates/ Accuracy

a. The drive system shall be capable of providing TBD$^{\circ}$/minute maximum tracking and TBD$^{\circ}$/minute minimum slew rates.

b. The drive assembly shall position the antenna mounting surface with respect to the gimbal assembly mounting surface to within TBD$^{\circ}$ throughout the entire operating range.

3.2.2.3.8 Tracking and Data Relay Antenna

A. Stowed Configuration. The 8-foot diameter, boom-mounted TDRSS antenna shall be furled and stowed within the Instrument Section Structure as shown in Figure 3.2.2-20. The overall package size shall be limited by the 86-inch diameter delta shroud.

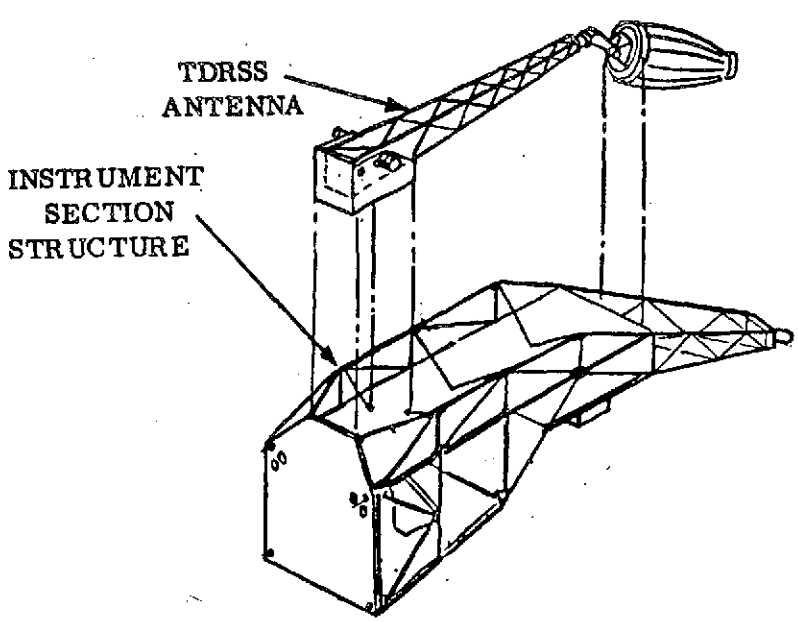


Figure 3.2-20. TDRSS Stow Configuration

B. Deployment. The TDRSS Antenna and boom shall be released from their stowed position, the boom erected, and the antenna unfurled via DC motorized drive. The deployed configuration shall be as shown in Figure 3.1-1. The operating mechanism shall be reversible so that the antenna may be furled on command and the boom returned into the stowed position.

C. Gimbal Rates/Accuracy. The drive system shall be capable of steering the TDRSS antenna over $\pm$TBD^{o} at TBD^{o}/min. maximum tracking rate and TBD^{o}/min. minimum slew rates. The antenna mounting surface shall remain within 0.3^{o} of the gimbal mounting surface for the reference EOS design.

3.2.2.3.9 Accommodations for the MPSS SPS Components

The Propulsion Module Structure, of the GPSS, shall include additional bracketry as required for the installation and support of the MPSS-designate SPS tanks, thrusters, and associated components.

3.2.2.4 Weight

The total deliverable weight of this structural subsystem shall not exceed 545 pounds allocated as follows:

GPSS - 360 lbs

MPSS - 185 lbs

3.2.2.5 Solar Array Shadowing

The solar array support structure shall be designed and located such that there shall be no shadowing of the solar array cells by the antenna or spacecraft body appendages when the array is deployed in orbit.

3.2.2.6 Sensor Fields of View

Design of the spacecraft shall be such that there will be no blockage of the sensor fields of view shown in Figure 3.2-21, by the array, antenna, or other spacecraft appendages during launch transfer and on-orbit operation.

3.2.3 RELIABILITY

In the presence of all identifiable degradation factors, the Structure Subsystem shall have a mission success probability of TBD for TBD years of operation allocated as follows:

Structural Performance	1.0
Solar Array Release and Deployment	TBD
Solar Array Drive	0.9999

Redundancy shall be provided in the mechanisms design to insure compliance with the stated life and reliability requirements.

3.2.4 MAINTAINABILITY

3.2.4.1 General Requirements

This subsystem shall be designed for ease of maintainability to minimize equipment down time during assembly, test, and checkout.

3.2.4.2 Maintenance and Repair Cycles

This subsystem shall be designed such that no scheduled maintenance will be required. Repairs shall be limited to factory repairs of structural components and thermal coatings and replacement or refurbishment of mechanism components prior to shipment.

3.2.4.3 Service and Access

Access shall be provided to individual equipments for inspection, servicing, and replacement without major disassembly of the spacecraft. Such access shall be accomplished by at most the removal of structural access panels. Special

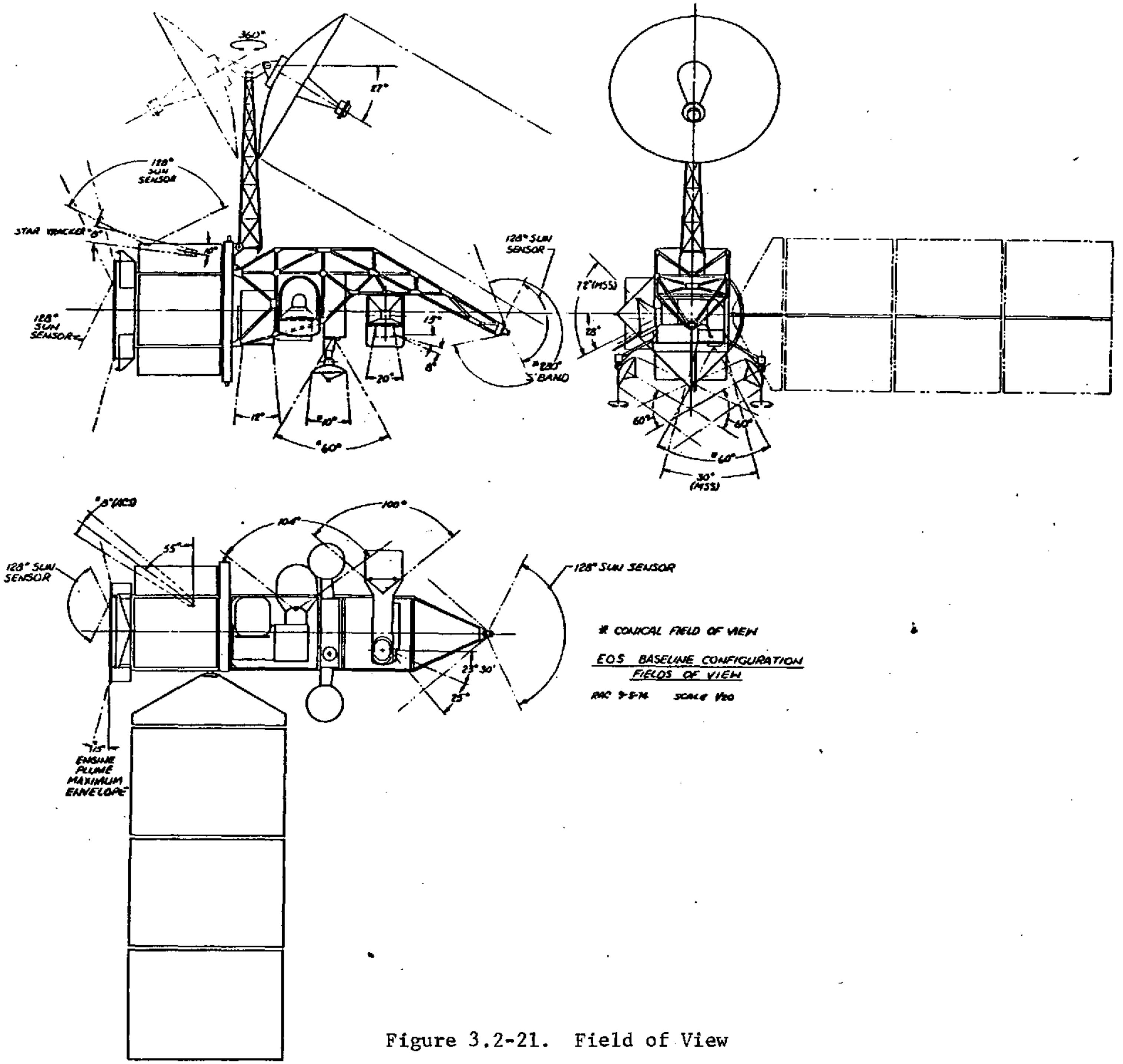


Figure 3.2-21. Field of View

equipments such as propulsion tanks, pressure vessels, tubing and valves, interconnected by welded, brazed or bonded joints need not be replaced except by disassembly of the spacecraft and/or the subsystem. Access shall be provided for installation and checkout of ordnance devices with the spacecraft located at the Western Test Range.

3.2.4.4 Handling and Assembly

The Structure Subsystem shall provide hard points for handling the subsystem, its subassemblies and the spacecraft. Hoist points, MAGE pick-ups and other handling provisions or restrictions shall be suitably marked by decals or other markings. Spacecraft major axis and/or station plane shall be identified for use of assembly. During handling operations all access doors must be secured and the solar array and antenna must be constrained in the stowed configuration.

3.2.4.5 Protective Covers

Removable dust covers or other protective devices shall be provided over electrical connectors. These devices shall be easily removed.

3.2.4.6 Replacement of Seals, Lubricants and Fluids

The useful life of the subsystem may be extended by the replacement of seals, lubricants and fluids in certain components and subsequent retest of the components.

3.2.5 ENVIRONMENTAL CONDITIONS

The subsystem structural elements shall be designed to withstand the environments and test conditions specified in TBD for the launch and orbital mission phases. The subsystem shall be capable of withstanding the transportation, storage and prelaunch environmental conditions of TBD subject to the interpretations of paragraphs 3.2.5.1 and 3.2.5.2.

The solar array panel structure and mechanism components of this subsystem shall be designed to withstand the environmental conditions specified in TBD.

The Structure Subsystem shall be operable during all mission phase of prelaunch, launch and orbit. The subsystem shall not operate during transport or storage conditions.

3.2.5.1 Transport, Handling and Storage

3.2.5.1.1 Assembled Spacecraft

The spacecraft, when transported, handled or stored in the assembled configuration, is supported or contained by its mechanical AGE in such a manner that the environments specified in paragraphs 6.1 and 6.2 of TBD are attenuated less than the critical flight environment such that the structure is not critical for this environment.

3.2.5.1.2 Major Subassemblies

The Structure Subsystem assemblies (without installed components) shall be capable of withstanding the temperature and pressure environments for storage and transportation of Table 1 of TBD in an unsheltered environment. Assemblies and subassemblies shall be protected from humidity and shock levels shall be attenuated to less than the critical flight environment.

3.2.5.2 Pre-Launch

The Structure Subsystem as part of the assembled spacecraft shall be encapsulated within the launch vehicle shroud at the Western Test Range. The spacecraft shall, therefore, be maintained in a sheltered environment during the pre-launch phase and exposed to the environments provided by conditioned air within the launch vehicle shroud as defined in TBD.

3.3 DESIGN AND CONSTRUCTION

3.3.1 MATERIALS, PROCESSES, AND PARTS

3.3.1.1 Selection of Materials, Processes and Parts

Materials and processes shall be selected in accordance with the requirements of (TBD) paragraph TBD.

3.3.1.2 Selection of Electronic Parts

Electronic parts shall be selected in accordance with the requirements of TBD, paragraph TBD.

3.3.1.3 Screening of Parts

Parts shall be screened in accordance with the requirements of TBD, paragraph TBD.

3.3.1.4 Parts Specifications

Parts specifications shall be prepared in accordance with TBD, paragraph TBD.

3.3.1.5 Part Application Restrictions

The application restrictions defined in TBD, paragraph TBD, shall apply.

3.3.1.6 Parts Derating

Parts shall be derated in accordance with TBD, paragraph TBD.

3.3.1.7 Traceability of Parts

The manufacturer's part number, lot number, and date code of all electronic parts assembled into prototype or flight equipment shall be recorded. Each part assembled to printed circuit boards shall have its identification markings visible after assembly.

3.3.1.8 Corrosion Prevention

The use of dissimilar metals as specified in MIL-STD-454, Requirement 16, shall be avoided wherever possible. Materials, techniques, and processes shall be selected and employed with regard to heat treatment procedure, corrosion protection, finish, and assembly and installation such that sustained or residual surface tensile stress, stress concentrations, and the hazards of stress corrosion, cracking, and hydrogen embrittlement are minimized. Processes and materials for protection against corrosion of metal parts shall be selected from those specified in paragraph 3.3.1.1, with the exception that cadmium plating shall not be used. Selected finishes shall be compatible with the thermal requirements of this specification. All parts shall be corrosion resistant or have a suitable protective coating applied.

3.3.1.9 Moisture and Fungus Resistance

Materials which are not nutrients for fungus and which resist damage from moisture shall be used wherever possible. The requirements of MIL-STD-454, Requirement 4 shall apply. The use of materials which are nutrients for fungus are not prohibited in hermetically sealed assemblies and in other accepted and qualified uses, such as paper capacitors and treated transformers. If it is necessary to use fungus nutrient materials in other than such qualified applications, these materials shall be treated with a process which will render the resulting exposed surface fungus resistant.

Protective coatings shall not be acceptable as moisture and fungus preventatives for parts which may lose their coatings during normal course of assembly, inspection, maintenance and testing.

3.3.2 ELECTROMAGNETIC COMPATIBILITY

Electromagnetic compatibility shall be in accordance with the detailed requirements of GE Specification SVS-________, Electromagnetic Compatibility Requirements for Components and Subsystems.

3.3.3 NAMEPLATES AND PRODUCT MARKING

a. The component of the subsystem shall be marked for identification in accordance with the manufacturer's standards. The identification shall include, but not be limited to, the following:

1. Nomenclature
2. Customer Part Number
3. Serial Number (Engineering models will use a different designation than prime hardware)
4. Contract Number
5. Manufacturer's Name or Trademark
6. Date of Manufacture (month, day, year)
7. Property:

b. Hardware or equipment which is not suitable for use in flight, and which would be accidentally substituted for Flight or Flight Spares Hardware shall be red striped with material compatible red paint to prevent such substitution. In the event the hardware is too small to be easily striped, or if test results would be affected by striping, a conspicuous red tag marked "NOT FOR FLIGHT USE" shall be attached.

c. Wires and Cables. Wires and cables for hardware shall not be identified by hot stamping directly onto primary or secondary (shield) insulation.

3.3.4 WORKMANSHIP

The subsystem, including all parts and subassemblies, shall be constructed, finished, and assembled in accordance with the highest standards for high reliability aerospace equipment. Workmanship criteria shall comply with MIL-STD-454, Requirements 9 and 24. Particular attention shall be paid to neatness and thoroughness of soldering, wiring, marking of parts and assemblies, plating, painting, machine screw assemblage and freedom from burrs and sharp edges.

3.3.5 CLEANLINESS

Hardware shall be designed, manufactured, assembled, and handled in a manner to insure the highest practical level of cleanliness. Suitable precautions shall be taken to insure freedom from debris within the hardware, and inaccessible areas where debris and foreign material can become lodged, trapped, or hidden shall be avoided. Hardware shall be designed so that malfunctions or inadvertent operating cannot be caused by exposure to conducting or nonconducting debris or foreign material floating in a gravity-free state. Electrical circuitry shall be designed and fabricated to prevent unwanted current paths being produced by such debris. Ultrasonic vibration shall not be used as a method for cleaning component electronic assemblies.

3.3.6 INTERCHANGEABILITY

Unless exception is taken and approved, the components, subassemblies and assemblies of the subsystem shall be interchangeable with regard to form, fit, and function with other components, subassemblies and assemblies of the same part number. Likewise, the components, subassemblies and assemblies themselves shall be directly interchangeable with other serial number components, subassemblies and assemblies. The requirements of MIL-STD-454, Requirement 7, shall apply.

3.3.7 SAFETY PRECAUTIONS

Warnings and precautions relative to personnel and equipment safety shall be specified in component handling, assembly and test instructions.

3.3.7.1 Personnel Safety

Personnel safety shall be in accordance with MIL-STD-454, Requirement 1. Adequate means for preventing inadvertent deployment of the solar panels during handling and test operation shall be employed.

3.3.7.2 Explosive and Ordnance Safety

Shorting caps shall be used to maintain a short circuit across each ordnance device during assembly and ground handling.

3.3.8 DRAWINGS

Drawing practices and procedures for the subsystem shall be compatible with the requirements of MIL-D-1000 and the standards of MIL-STD-100.

3.3.9 WELDING

Structural welds shall comply with MIL-W-8604 and MIL-W-8658. Brazing shall be in accordance with MIL-B-7883.

SECTION 4

VERIFICATION

4.1 GENERAL

In general, tests and inspections performed to verify the requirements specified in Section 3 shall be accomplished at the General Electric Company, Valley Forge Space Center. However, certain components designed by subcontractors shall be inspected and tested by the subcontractor.

4.2 VERIFICATION METHODS

Table 4.2-1, Requirements Verification Matrix, shall define the method of verification for each requirement specified in Section 3. Verification methods, and requirements for their use, are defined in subparagraphs below.

Table 4.2-2, Environmental Test Matrix, shall define the requirements of Section 3 that are to be verified in the specified qualification or acceptance environment.

4.2.1 SIMILARITY

Verification testing or analysis may be waived if it can be shown through analysis of existing records that the item is similar or identical in design and manufacturing processes to another item that has previously been qualified to equivalent or more stringent criteria. Similarity may pertain to characteristics such as material, configuration, and functional element or assembly, and may be applied selectively for applicable environments.

4.2.2 ANALYSIS

Analytical techniques may be used, in lieu of testing to verify compliance to specified requirements, either alone or in combination with test. The selected techniques may include, typically, system engineering analysis, statistics, qualitative analysis, analog modeling, and computer simulation.

Table 4.2-1. Requirements Verification Matrix

Section 3 Paragraph		N/A	Verification Method: Similarity 4.3	Inspection 4.4	Analysis 4.5	4.6 Test: Qual 4.6.1	Accept 4.6.2
3.1	Item Definition	X					
3.1.1	Item Description	X					
3.1.1.1	GPSS	X					
3.1.1.2	MPSS	X					
3.1.2	Interface Definition	X					
3.1.2.1	GPSS/MPSS Interface	X					
3.1.2.2	Launch Vehicle Interface				X.1		
3.1.2.2.1	Delta 2910 Interface				X.1		
3.1.2.2.2	Titan IIIB Interface				X.1		
3.1.2.3	Space Shuttle Interface				X.1		
3.1.2.3.1	Transition Frame						
3.1.2.3.2	Handling Provisions						
3.1.2.4	Mechanical Interfaces	X					
3.1.2.4.1	LV Adapter/SPS Module		Sample Matrix				
3.1.2.4.2	SPS						
3.1.2.4.3	SPS Module/S/S Section		Final Matrix TBD				
3.1.2.4.4	S/S Section/S/S Module						
3.1.2.4.5	S/S Section/Transition Fr.						
3.1.2.4.6	Transition Frame/IS Struct.						
3.1.2.4.7	IS Struct./Mission Payloads						
3.1.2.4.8	S/S Section/SA Support				X.1		
3.1.2.4.9	SA Support Structure				X.1		
3.1.2.4.10	Component Installation				X.1		
3.1.2.4.11	Electrical Harness				X.1		
3.1.2.4.12	Thermal Blankets				X.1		
3.1.2.5	Electrical Interfaces					X.3	X.3
3.1.2.5.1	Electro-Explosive Devices					X.3	X.3
3.1.2.5.2	Verification Switches					X.3	X.3
3.1.2.6	Thermal Interfaces	X					
3.1.2.6.1	Insulation,Coatings,Finishes				X.1		
3.1.2.6.2	Conductance				X.1		
3.1.2.6.3	Surface Flatness				X.1		
3.1.2.6.4	Alignment				X.1		
3.1.2.7	GSE Interfaces	X					
3.2	Characteristics	X					
3.2.1	Performance	X					
3.2.1.1	Functional Performance	X					
3.2.1.2	Struct. Performance	X					
3.2.1.2.1	Support				X.2		
3.2.1.2.2	Stiffness						
A	Powered Flight						
B	Orbit		Sample Matrix				
3.2.1.2.3	Strength						
A	Critical Load Conditions		Final Matrix TBD				

Table 4.2-1. Requirements Verification Matrix

Section 3 Paragraph		N/A	Verification Method: Similarity 4.3	Inspection 4.4	Analysis 4.5	4.6 Test: Qual 4.6.1	Accept 4.6.2
B	Shock						
C	Ground Handling, Transportation						
D	Steady State Accelerations				X.2		
E	Factors of Safety				X.2		
F	Margins of Safety				X.2		
3.2.1.2.4	Alignment				X.2		
3.2.1.3	Mechanical Functions	X					
3.2.1.3.1	Module Latch Mechanisms					X.1	X.1
3.2.1.3.2	Launch Vehicle Sep. Device					X.1	X.1
3.2.1.3.3	Solar Array Retention/ Deployment	X					
A	Retention					X.1	X.1
B	Release/Deployment/Retraction	X					
a	Contamination				X.3		
b	Deployment/Retraction Time				X.3		
c	Deployment/Retraction Forces				X.3		
d	Position & Clearances				X.3		
c	Stops, Latches, & Releases	X					
3.2.1.3.4	Solar Array Drive & Power Transfer	X					
3.2.1.3.5	Wideband antenna	X					
3.2.1.3.6	TDRSS Antenna	X					
3.2.1.4	Useful Life				X.4		
3.2.2	Physical Characteristics	X					
3.2.2.1	Configuration	X					
3.2.2.2	GPSS						
3.2.2.2.1	Subsystem Section Struct.		Sample Matrix				
3.2.2.2.2	Propulsion Module Struct.						
3.2.2.2.3	S/S Module Substructure		Final Matrix TBD				
3.2.2.2.4	Transition Frame						
3.2.2.2.5	Module Latching Mech.						
3.2.2.3	MPSS	X					
3.2.2.3.1	Launch Vehicle Adapter	X					
3.2.2.3.2	Solar Array Ass'y & Drive Mech.	X					
3.2.2.3.3	Instrument Section Struct.	X					
3.2.2.3.4	Launch Vehicle Separation	X					
A	Vee-band					X.1	X.1
B	Separation Springs					X.1	X.1
3.2.2.3.5	Solar Array Deployment/ Retraction	X					
A	Synchronized Motion					X.1	X.1
B	Latching/Unlatching					X.1	X.1

Table 4.2-1. Requirements Verification Matrix

Section 3 Paragraph		N/A	Verification Method				
			Similarity	Inspection	Analysis	4.6 Test	
						Qual	Accept
			4.3	4.4	4.5	4.6.1	4.6.2
3.2.2.3.6	Solar Array Drive					X.1	X.1
3.2.2.3.7	Wide Band Antenna	X					
A	Configuration	X					
B	Gimbel Rates/Accuracy					X.1	X.1
3.2.2.3.8	TDRSS Antenna	X					
A	Stowed Configuration	X					
B	Deployment					X.1	X.1
C	Gimbal Rates/Accuracy						
3.2.2.3.9	SPS Component Mounting						
3.2.2.4	Weight		Sample	Matrix			
3.2.2.5	Solar Array Shadowing						
3.2.2.6	Sensor Fields of View		Final	Matrix	TBD		
3.2.3	Reliability				X.4		
3.2.4	Maintainability	X					
3.2.4.1	General Requirements				X.5		
3.2.4.2	Maintenance & Repair Cycles				X.5		
3.2.4.3	Service and Access				X.5		
3.2.4.4	Handling and Assembly				X.5		
3.2.4.5	Protective Covers				X.5		
3.2.4.6	Replacement of Seals, Lubricants and Fluids				X.5		
3.2.5	Environmental Conditions					X.1	X.2
3.2.6	Transportability				X.6		
3.3	Design and Construction	X					
3.3.1	Materials,Processes, & Parts				X.7		
3.3.1.1	Selection of Materials, Processes, and Parts				X.7		
3.3.1.2	Selection of Elec.Parts						
3.3.1.3	Screening of Parts		Sample	Matrix			
3.3.1.4	Parts Specifications						
3.3.1.5	Part Application Restrictions						
3.3.1.6	Parts Derating		Final	Matrix	TBD		
3.3.1.7	Traceability of Parts						
3.3.1.8	Corrision Prevention						
3.3.1.9	Moisture & Fungus Resis.						
3.3.2	Electromagnetic Compatibility				X.8		
3.3.3	Nameplates & Prod.Marking			X			
3.3.4	Workmanship			X			
3.3.5	Cleanliness			X			
3.3.6	Interchangeability						
3.3.7	Safety Precautions	X					

Table 4.2-1. Requirements Verification Matrix

Section 3 Paragraph		N/A	Verification Method: Similarity 4.3	Inspection 4.4	Analysis 4.5	4.6 Test: Qual 4.6.1	Accept 4.6.2
3.3.7.1	Personnel Safety				X.9		
3.3.7.2	Explosive & Ordnance Safety				X.9		
3.3.8	Drawings				X.1		
3.3.9	Welding and Brazing				X.1		

Table 4.2-2. Environmental Test Matrix

Section 3 Paragraph Number \ Paragraph Number and Title	QUALIFICATION										ACCEPTANCE				
	Perform	Oper Temp	Sine Vib	Random Fib	Shock	Acoustic	Accel	Thermal Balance	Thermal Vacuum	Thermal Ambient	Performance	Sine Vib	Random Vib	Acoustic	Thermal Vacuum
	4.1	4.2.1	4.2.2.1	4.2.2.2	4.2.3	4.2.4	4.2.5	4.2.6	4.2.7	4.2.8	4.1	4.3.1.1	4.3.1.2	4.3.2	4.3.3
3.1.2.5	X										X				
3.1.2.5.1	X										X				
3.2.1.2.1			X	X			X								
3.2.1.2.2	X														
3.2.1.2.3							X								
3.2.1.2.3 A	X		X	X			X		X		X				
3.2.1.2.3 B	X		X	X					X		X				
3.2.2.4	X										X				

(SAMPLE MATRIX - FINAL MATRIX TBD)

4.2.3 INSPECTION

Inspection may be used to verify design and construction requirements, drawing compliance, or specific physical dimensions of the item.

4.2.4 TEST

Test may be used as the verification method either alone or in combination with analysis. Applicable tests may be: functional/performance including mechanical, electrical, electromechanical, environmental, life, alignment, weight and center of gravity, etc. Performance and functional tests are defined as:

- <u>Performance Test</u>. Quantitative tests that verify requirements within specified limits.
- <u>Functional Test</u>. Qualitative tests that verify operation. Quantitative measurements may be performed.

In general, performance tests are required for qualification and functional tests are required for acceptance. Performance tests shall be conducted on selected specified parameters as a basis for acceptance.

4.3 <u>VERIFICATION BY SIMILARITY</u> - None.

4.4 <u>VERIFICATION BY INSPECTION</u>

Visual inspection of the as-built subsystem shall be used to verify for the requirements so designated in Table 4.2-1.

4.5 <u>VERIFICATION BY ANALYSIS</u>

4.5.1 DRAWING REVIEW AND EVALUATION

Drawings prepared for the structural subsystem shall be reviewed and evaluated for compliance with the mechanical interface requirements of paragraph 3.1.2 of this specification. In addition, a partial scale spacecraft configuration layout to verify the overall configuration, sensor field-of-view clearances and sun shadow restrictions shall be prepared.

4.5.2 STRUCTURAL ANALYSES

Stress and dynamic structural analyses shall be made to verify the structural support, stiffness and strength requirements which cannot be directly verified by test.

4.5.3 DYNAMIC MOTION ANALYSES

The design of Solar Array Assembly and the TDRSS shall be applied to dynamic motion analyses to verify certain requirements which cannot be readily measured by test in a 1g environment.

4.5.4 LIFE AND RELIABILITY ANALYSIS

Standard analyses techniques shall be used to verify life and reliability requirements.

4.5.5 MAINTAINABILITY

Drawings prepared for the subsystem shall be reviewed for compatibility with maintainability requirements. In addition, the actual use of development and protoflight hardware shall be analyzed by maintainability verification.

4.5.6 TRANSPORTABILITY ANALYSIS

Load factors imposed by the modes of transportation shall be compared to launch load factors to evaluate the structural capability during transportation. Shipping lists and instructions shall be reviewed to evaluate compatibility with pyrotechnic device requirements.

4.5.7 MATERIALS, PROCESSES, AND PARTS SELECTION

The subtier specifications and drawings prepared for the subsystem shall be reviewed and evaluated for compatibility with the requirements of paragraph 3.3.1.

4.5.8 ELECTROMAGNETIC COMPATIBILITY

(TBD)

4.5.9 SAFETY ANALYSIS

Drawings, handling procedures, assembly and test instructions, and shipping instructions shall be reviewed for compatibility with safety requirements of paragraph 3.3.7.

4.6 VERIFICATION BY TEST

4.6.1 QUALIFICATION TEST

4.6.1.1 Structures Development Model

The structure subsystem shall be qualified by test of an engineering model of the subsystem called the Structures Development Model (SDM). The SDM shall be a full scale model of the spacecraft with structure representative of the final design. The structural elements used in the SDM shall be inspected and certifications of the materials used shall be provided. Other spacecraft equipments shall be mass simulated in the SDM.

Tests to be performed with the SDM shall include:

- o Solar array deployment and retraction at the assembly level and at the spacecraft level
- o TDRSS deployment and retraction at the assembly level and at the spacecraft level
- o Sinusoidal vibration surveys
- o Sinusoidal and random vibration to spacecraft qualification test levels
- o Pyrotechnic shock by firing of actual pyrotechnic devices
- o Static load tests to simulate limit steady-state accelerations
- o Alignment checks in between various environmental tests

4.6.1.2 Solar Array Assembly and Drive Mechanism (SAADM) Protoflight Model

The SAADM unit fabricated for the protoflight spacecraft shall be qualification tested at the assembly level.

4.6.1.3 Tracking and Data Relay Subsystem (TDRSS)

The TDRSS unit fabricated for the protoflight spacecraft shall be qualification tested at the assembly level.

4.6.1.4 Pyrotechnics

Pyrotechnic devices shall be qualified as a lot quantity. Electrical interfaces for pyrotechnic devices shall be checked at the spacecraft level of assembly by wiring checkout and actual firing tests.

4.6.1.5 Weight Measurement

The weight of the various components, assemblies and subassemblies shall be measured and recorded.

4.6.2 ACCEPTANCE TEST

4.6.2.1 SAADM Acceptance Test

Acceptance tests shall be performed at the SAADM level of assembly prior to installation on the spacecraft.

4.6.2.2 TDRSS Acceptance Test

Acceptance tests shall be performed at the TDRSS level of assembly prior to installation on the spacecraft.

4.6.2.3 Electrical Interfaces

Electrical interfaces shall be verified at the spacecraft level of assembly.

4.6.2.4 Weight Measurement

The weight of the various components, assemblies and subassemblies shall be measured and recorded.

SECTION 4.0

Specification SVS-XXXX
September 16, 1974

Specification for the

EARTH OBSERVATORY SATELLITE THERMAL CONTROL SUBSYSTEM

TABLE OF CONTENTS

SECTION 1

SCOPE

This specification establishes the performance, design, development, and test requirements for the Thermal Control Subsystem prime item, hereafter referred to as the TCS.

SECTION 2

APPLICABLE DOCUMENTS

The following documents of the exact issue shown form a part of this specification to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

2.1 GENERAL ELECTRIC DOCUMENTS

2.1.1 DRAWINGS

(TBD) Launch Vehicle/Spacecraft Mechanical Interface

(TBD) TCS Interface Drawing - EOS

(TBD) TCS Electrical Schematic

(TBD) TCS Installation

(TBD) Heater, Foil

(TBD) Electronic Thermostat

2.1.2 SPECIFICATIONS

(TBD) Electrical System Interface for EOS

(TBD) Specification for Thermostat for EOS

(TBD) System Specification of the EOS

(TBD) Environmental Test Requirements for Components, Subsystems and Payloads of EOS

(TBD) Electromagnetic Compatibility Requirements for Components and Subsystems for EOS

(TBD) EOS Approved Materials and Process List

(TBD) EOS Approved Parts List

2.2 U.S. GOVERNMENT DOCUMENTS

MIL-STD-454D 31 Aug. '73	Standard General Requirements for Electronics Equipment

2.3 OTHER

DSDR 61687	Delta Spacecraft Design Restraints

SECTION 3

REQUIREMENTS

3.1 THERMAL CONTROL SUBSYSTEM DEFINITION

3.1.1 THERMAL CONTROL SUBSYSTEM DESCRIPTION

The thermal control subsystem, hereinafter called the TCS, shall consist of thermal control coatings, insulation, heaters, temperature sensors and thermostats. The TCS shall maintain all required spacecraft, subsystem, and component temperature levels and gradients during launch, and throughout the operational orbital lifetime. This temperature control shall be maintained by semi-passive means employing existing space-flight qualified components and materials.

3.1.1.1 TCS Block Diagram

The TCS block diagram defining the basic functional flow schematic with TCS component characteristics is shown in Figure 3-1.

3.1.2 INTERFACE DEFINITION

3.1.2.1 External Interfaces

3.1.2.1.1 Fairing

3.1.2.1.1.1 Surface Emissivity

The fairing external surface shall have a solar absorptance which is less than or equal to 0.35 (TBR).

3.1.2.1.1.2 Fairing Blanket

The fairing shall be fitted with an acoustic blanket which shall have a thermal conductivity of 0.05 BTU/hr-ft-°F or less except at the retaining channels (TBR).

3.1.2.1.2 Launch Vehicle

The TCS interface with the launch vehicle shall be as defined in General Electric Drawing (TBD) , Launch Vehicle/spacecraft Mechanical Interface.

3.1.2.1.3 Spacecraft Mechanical Support Equipment (MSE) - Launch Operations Support

The TCS interface with the Spacecraft MSE-Launch Operations Support shall be as defined in General Electric Drawing (TBD) , Launch Vehicle/Spacecraft Mechanical Interface.

3.1.2.2 <u>Internal Interfaces</u>

3.1.2.2.1 TCS Component Interfaces

The interfaces between TCS components shall be as defined in Figure 3-1.

3.1.2.2.2 ACS Module

3.1.2.2.2.1 <u>ACS Component Interface</u>

The TCS interface with all ACS components shall be as defined in Table 3-1.

3.1.2.2.2.2 <u>ACS Component Thermal Dissipation</u>

The thermal dissipation of each ACS electrical component shall be provided to the TCS as shown in Table 3-2.

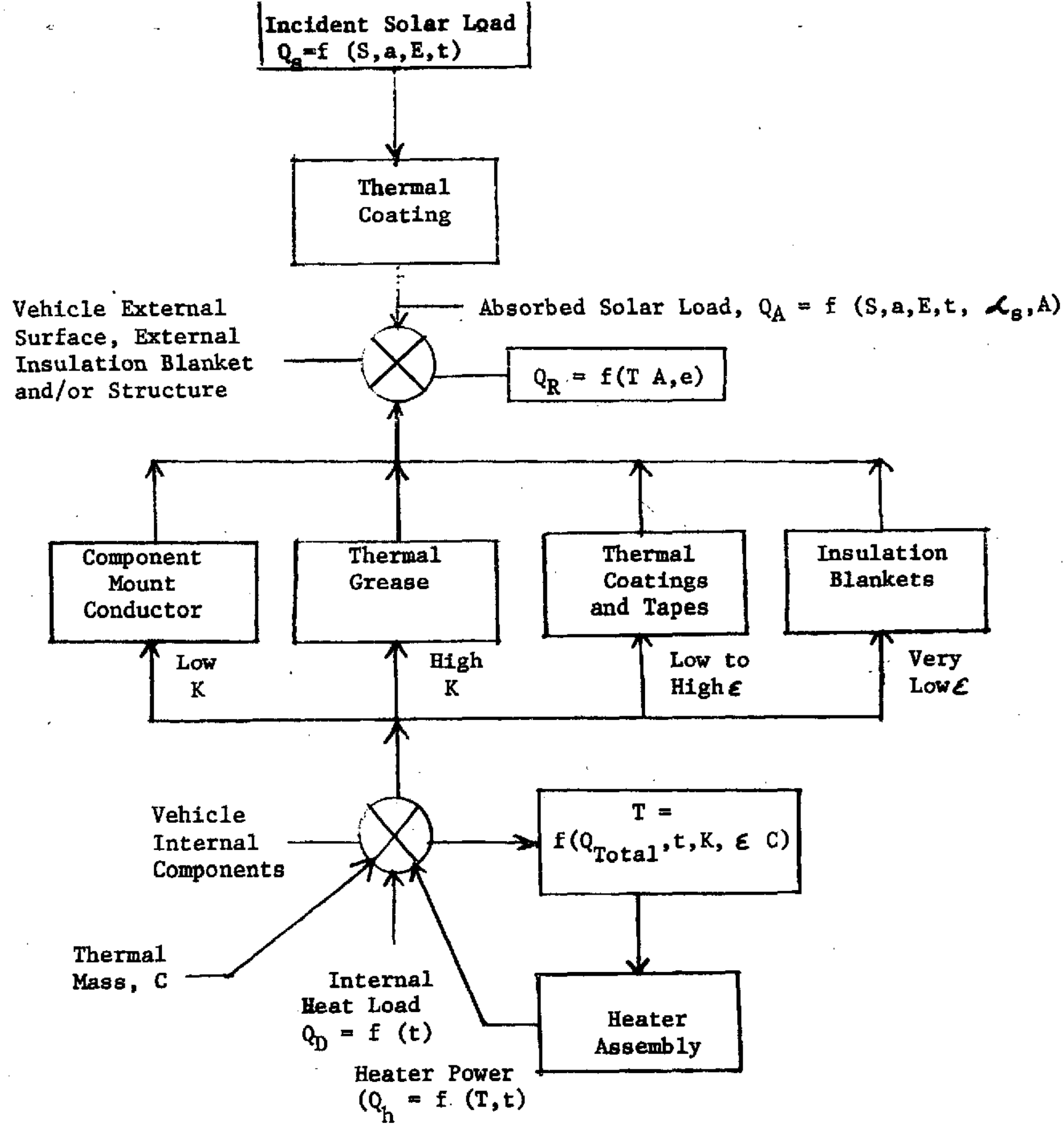
3.1.2.2.2.3 <u>Antenna and Feed Assembly</u>

The TCS interface with the Antenna Assembly shall be as defined in General Electric Drawing (TBD) , TCS Interface Drawing - BSE.

3.1.2.2.3 C&DH Module

3.1.2.2.3.1 <u>C&DH Component Interface</u>

The TCS interface with all C&DH components shall be as defined in Table 3-1.



Nomenclature

Q - Power
T - Temperature
S - Solar Const.
t - Time
A - Area
α_s - Solar absorptivity
ϵ - Hemispherical emissivity
K - Thermal Conductivity
C - Thermal Capacitance
A - Area
a - Albeda Flux
E - Earth Flux

Figure 3-1. Thermal Control Subsystem Functional Schematic

3.1.2.2.3.2 C&DH Component Thermal Dissipation

The thermal dissipation of each C&DH electrical components shall be provided to the TCS as shown in Table 3-2.

3.1.2.2.4 Power Module

3.1.2.2.4.1 TCS Component Power Source

All TCS components requiring electrical interfaces shall be designed to operate from a power source having the characteristics specified in the subparagraphs of paragraphs (TBD) and (TBD) in Specification (TBD) .

3.1.2.2.4.2 Harness Interface

The TCS interface with the electrical harness shall be as defined on General Electrid Drawing (TBD) , TCS Electrical Schematic.

3.1.2.2.4.3 Electrical Short

The Power Module shall protect the vehicle electrical system and all other TCS heater assemblies from a short in any one heater assembly, or combination of heater assemblies.

3.1.2.2.4.4 Command Capability

The Power Module shall provide the TCS heater assemblies with the command capability shown in Table 3-4.

3.1.2.2.4.5 Power Component Interface

The TCS interface with all Power Module components shall be as defined in Table 3-1.

3.1.2.2.4.6 Power Component Thermal Dissipation

The thermal dissipation of each Power Module electrical component shall be provided to the TCS as shown in Table 3-2.

3.1.2.2.4.7 Thermostat Electrical Interface

Table 3-1. Component Thermal Interface (TBR) Requirements

Subsystem/Component	Finish	Flatness	Grease	Mounting Bolt Torque	Surface Emissivity*
ACS Module					
Inertial Reference Unit	64	.005	Yes	25	$\geq$.80
Star Detector/Sensor	TBD	TBD	TBD	25	$\geq$.80
Magnetic Unload Unit	64	.005	Yes	25	$\geq$.80
Momentum Wheels	64	.005	Yes	25	$\geq$.80
Sun Sensor	TBD	TBD	TBD	25	$\geq$.80
Actuator Drive Unit	64	.005	Yes	25	$\geq$.80
Magnetometer	TBD	TBD	TBD	25	TBD
Power Regulation Unit	64	.005	Yes	25	$\geq$.80
C&DH Module					
S-Band Transponder	64	.005	Yes	25	$\geq$.80
Modulator/Demodulator					
Central Command Decoder					
Telemetry Format Generator					
Clock & Time Code Gen.					
Command Decode Remotes					
Telemetry Mux Remotes					
OBC (On-Board Computer)					
NBTR					
Power Regulator					
DCS	64	.005	Yes	25	$\geq$.80
Power Module					
Central Control Unit	64	.005	Yes	25	$\geq$.80
Power Reg. Unit #1					
Power Reg. Unit #2					
Power Reg. Unit #3					
Battery #1					
Battery #2					
Battery #3					
Power Control Unit					
Remote Decoder (2)					
Remote Mux. (2)					
S/C Int. Ass'y.					
Test Conn. Ass'y.	64	.005	Yes	25	$\geq$.80
Low Thrust Engine	N/A	N/A	No	TBR	$\leq$0.15
Medium Thrust Engine					
High Thrust Engine					
Fill and Vent Valve					
Fill and Drain Valve for Hydrazine					
Pressure Transducer					
Filter					
Latching Valve					
Tanks					
Lines					
Valves	N/A	N/A	No	TBR	$\leq$0.15
Solar Array Drive	N/A	N/A	N/A	N/A	TBR

Table 3-2. Temperature Dissipation Limits and Power (TBR)

Module/Component	Design Temp. Limits (°F)	Thermal Diss. (Watts) Max.	Min.
<u>Mission Independent</u>			
ACS Module	70 ± 5	105.6	86.8
Power Module	50 ± 5	113.5	92.9
C&DH Module	70 ± 5	153.7	125.7
RCS Tanks			
Lines & Valves	40 to 120	-	-
Engine Catalyst Bed (Prior to Firing)	250 min.	-	-
<u>Mission Peculiar</u>			
Solar Array Drive	30 to 110	1.8	1.2
Solar Array	-85 to 149	-	-
Wide Band Module	70 ± 5	TBR	TBR
Thematic Mapper Electronics			
Electronics	70 ± 5	TBR	TBR
Cooler	-280 to -320	.005	.0
MSS			
Electronics	70 ± 5	TBR	TBR
Cooler	-280 to -320	.005	.0
TDRSS Antenna	TBR	TBR	TBR
Propulsion Tanks			
Lines & Valves	40 to 120	-	-
Engine Catalyst Bed (Prior to Firing)	250 min	-	-

The interface of the thermostats with the power controller and heaters shall be as defined in General Electric Specification (TBD), paragraph (TBD) .

3.1.2.2.4.8 Flight Temperature Sensors

The TCS shall be provided with the flight temperature sensors shown in Table 3-3. The temperature sensor accuracy shall be within TBD of the sensor temperature range.

3.1.2.2.5 Propulsion Module

The TCS interface with the Propulsion Module shall be as defined in General Electric Drawing TBD, Thermal Control Subsystem Interface Drawing - EOS.

3.1.2.2.6 Structure Subsystem (SS)

3.1.2.2.6.1 Ground Conditioning

Conditioned air shall be supplied to the spacecraft from the time of fairing installation until liftoff. The minimum air quantity shall be 1500 cubic feet per minute at a temperature of 57° $\pm$ 5°F (TBR).

3.1.2.2.6.2 TCS Installation

The TCS shall be attached to the spacecraft structure as shown in General Electric Drawing (TBD) , TCS Installation.

3.1.2.2.6.3 Flight Sensor Locations

The location of the flight temperature sensors allocated to the TCS shall be as shown in General Electric Drawing (TBD) , TCS Installation.

3.1.2.2.6.3.1 Sensor Bonding

Bonding material shall have a minimum volume resistivity of 0.01 ohm/cm, a minimum tensile sheer strength of 450 psi, and a minimum thermal conductivity of 40 BTu/hr. °F ft.

Table 3-3. Flight Sensor Requirements (TBR)

Location	Temperature Range	Number
Antenna	TBD	1
Antenna Feed		1
Tank Str. #1		1
Tank Str. #2		1
Latch Valve Assembly		1
Lines -Zones 1-6		6
Analogue Sun Sensor A Temp.		1
Analogue Sun Sensor B Temp.		1
Battery #1 Temp.		1
Battery #2 Temp.		1
Battery #3 Temp.		1
Top S/A Temp.		1
Bottom S/A Temp.		1
Tank Temp. #1		1
Tank Temp. #2		1
Thruster #1-14 Temps.		14
SAD Temp.		1
ACS Module		3
Power Module		3
C&DH Module		3
Wide Band Module		3
Thematic Mapper		3
MSS		3
Structure		6
Spares		5
		64

Table 3-4. Command Allocations (TBR)

1. Propulsion Valves Primary Heaters Enable
2. Propulsion Valves Back-up Heaters Enable
3. Propulsion Valves Heaters Off
4. Catalyst Bed Heaters (LTE 1--4) On
5. Catalyst Bed Heaters (LTE 5--8) On
6. Catalyst Bed Heaters (MTE 9 and 12) On
7. Catalyst Bed Heater (HTE 13) On
8. Catalyst Bed Heater (HTE 14) On
9. Catalyst Bed Heaters Off
10. ACS Module Primary Heater Enable
11. ACS Module Back-up Heater Enable
12. ACS Module Heaters Off
13. Power Module Primary Heater Enable
14. Power Module Back-up Heater Enable
15. Power Module Heaters Off
16. C&DH Module Primary Heater Enable
17. C&DH Module Back-up Heater Enable
18. C&DH Module Heaters Off
19. Wideband Module Primary Heater Enable
20. Wideband Module Back-up Heater Enable
21. Wideband Module Heaters Off
22. Thematic Mapper Primary Heater Enable
23. Thematic Mapper Back-up Heater Enable
24. Thematic Mapper Heaters Off
25. MSS Primary Heater Enable
26. MSS Back-up Heater Enable
27. MSS Heaters Off

3.1.2.2.6.4 Ascent Flight

3.1.2.2.6.4.1 Time-Temperature Profile - Fairing Acoustic Blanket

The time-temperature profile of the fairling acoustic blanket surface facing the spacecraft shall be as shown in Figure TBD and as noted in paragraph (TBD) of (TBD) .

3.1.2.2.7 Mission Peculiar Modules

3.1.2.2.7.1 Heat Exchange

The maximum heat loss or gain to the support structure from the wideband, thematic mapper, or MSS Modules shall be (TBD) watts.

3.1.2.2.7.2 Cryogenic Coolers

The thematic mapper and MSS cryogenic coolers shall be provided an unobstructed view of deep space. Radiation shields shall be provided as required.

3.2 CHARACTERISTICS

3.2.1 PERFORMANCE

3.2.1.1 General

The TCS shall have the general performance characteristics defined in paragraph (TBD) of General Electric Specification (TBD) .

3.2.1.2 Temperatures

The TCS shall maintain all spacecraft component operational temperature levels as defined in Table 3-2.

3.2.1.3 Thermal Dissipations

The TCS shall maintain the component temperatures specified in Section 3.2.1.2 with the component thermal dissipations as defined in Table 3-2.

3.2.1.4 Mission Phases

The TCS shall maintain the requirements of Section 3.2.1.2 while operating in the mission phases defined in paragraph (TBD) of General Electric Specification (TBD) .

3.2.1.5 Attitude Control

The TCS shall maintain the requirements of Section 3.2.1.2 while the spacecraft attitude control is as defined in paragraph (TBD) of General Electric Specification (TBD) .

3.2.1.6 Useful Life

The minimum TCS design useful life shall be as follows:

a. Shelf Life - 1 year after acceptance test

b. Operational Life in Orbit - 2 years

3.2.2 PHYSICAL CHARACTERISTICS

3.2.2.1 Installation

The TCS spacecraft installation shall be as defined in General Electric Drawing (TBD) , TCS Installation.

3.2.2.2 Size

The TCS space allocation shall be determined from the TCS component sizes defined in Table 3-5.

3.2.2.3 Weight

The TCS weight shall not exceed TBD pounds. The weight allocation for the individual TCS components is shown on Table 3-6.

3.2.2.4 Power

The maximum TCS heater power requirements by component/subsystem phase shall be as shown on Table 3-10.

3.2.2.5 Quantity

The quantity of TCS components required per spacecraft shall be as shown on Table 3-7.

3.2.2.6 Transportation and Storage

The TCS shall be transported and stored in the installed configuration as a part of the total assembled spacecraft, while subject to the environments specified in (TBD), Storage and Transportation (Packaged) columns of General Electric Specification (TBD).

3.2.3 RELIABILITY

The 2-year probability of the TCS meeting the performance requirements of this specification shall be TBD. This reliability is apportioned solely to the TCS heater/thermostat assemblies. No single failure shall cause loss of a mission critical function.

3.2.4 MAINTAINABILITY

The TCS shall contain no expendables or piece parts likely to age or wear out during the required life. No component shall require periodic maintenance or adjustment.

3.2.5 ENVIRONMENTAL CONDITIONS

With the TCS installed on the spacecraft its performance requirements as defined in this specification shall be satisfied when the spacecraft or its container has been exposed to the environmental conditions specified in paragraphs of General Electric Specifications (TBD) :

Table 3-5. Component Sizes (TBR)

Component	Maximum Volume
Insulation Blankets	.5" x area
Thermal Coatings	.01" x area
Thermal Grease	.01" x area
Stycast Conductors	TBR
Thermal Tapes	.01" x area
Thermal Fasteners	
o Brackets	TBD
o Nylon Thread	Included with blanket
o Velcro Strip	.1" x area
o Insulation Button	1" dia. x .75" high
o Adhesives	.01" x area
Heater Assembly	
o Heater	.02" x area
o Electronic Thermostat	3" x 2" x 1"

Table 3-6. Weight Allocation (TBR)

Component	Weight (#)
Insulation Blankets	46.0
Thermal Coatings*	3.7
Thermal Grease	0.50
Stycast Conductors	0.50
Thermal Tapes	1.6
Thermal Fasteners	9.2
Heater Assembly	
Heaters	1.6
Thermostat	4.2
Total	67.3

*Solar Array Coating Weight included in Power Subsystem

Table 3-7. TCS Quantity per Spacecraft (TBR)

Component	Number
Insulation Blankets (460 ft^2)	1 set
Thermal Coatings	1 set
Thermal Grease	A/R
Stycast Conductors	A/R
Thermal Tapes	A/R
Thermal Fasteners	1 set
o Brackets	
o Nylon Tread	
o Velcro Strip	
o Insulation Buttons	
o Adhesives	
Heater Assembly	
o Heater	58
o Thermostat	9

3.2.6 TRANSPORTABILITY

During transportation, the TCS shall be subject to transportation environments no more severe than those given in paragraph (TBD) of General Electric Specification (TBD) . The TCS shall be capable of meeting all performance requirements of Section 3 of this specification after being subjected to these transportation environments.

3.3 DESIGN AND CONSTRUCTION

3.3.1 MATERIALS, PROCESSES, AND PARTS

3.3.1.1 Selection of Materials, Processes and Parts

Materials and processes shall be selected in accordance with the requirements of (TBD) , paragraph (TBD) .

3.3.1.2 Selection of Electronic Parts

Electronic parts shall be selected in accordance with the requirements of (TBD) , paragraph (TBD) .

3.3.1.3 Screening of Parts

Parts shall be screened in accordance with the requirements of (TBD) , paragraph (TBD) .

3.3.1.4 Parts Specifications

Parts specifications shall be prepared in accordance with (TBD) , paragraph (TBD) .

3.3.1.5 Part Application Restrictions

The application restrictions defined in (TBD) , paragraph (TBD) , shall apply.

3.3.1.6 Parts Derating

Parts shall be derated in accordance with (TBD) , paragraph (TBD) .

3.3.1.7 Traceability of Parts

The manufacturer's part number, lot number, and date code of all electronic parts assembled into prototype or flight equipment shall be recorded. Each part assembled to printed circuit boards shall have its identification markings visible after assembly.

3.3.1.8 Corrosion Prevention

The use of dissimilar metals as specified in MIL-STD-454, Requirement 16, shall be avoided wherever possible. Materials, techniques, and processes shall be selected and employed with regard to heat treatment procedure, corrosion protection, finish, and assembly and installation such that sustained or residual surface tensile stress, stress concentrations, and the hazards of stress corrosion, cracking, and hydrogen embrittlement are minimized. Processes and materials for protection against corrosion of metal parts shall be selected from those specified in paragraph 3.3.1.1. Selected finishes shall be compatible with the thermal requirements of this specification. All parts shall be corrosion resistant or have a suitable protective coating applied.

3.3.1.9 Moisture and Fungus Resistance

Materials which are not nutrients for fungus and which resist damage from moisture shall be used wherever possible. The requirements of MIL-STD-454; Requirement 4, shall apply. The use of materials which are nutrients for fungus are not prohibited in hermetically sealed assemblies and in other accepted and qualified uses; such as paper capacitors and treated transformers. If it is necessary to use fungus nutrient materials in other than such qualified applications, these materials shall be treated with a process which will render the resulting exposed surface fungus resistant.

Protective coatings shall not be acceptable as moisture and fungus preventatives for parts which may lose their coatings during normal course of assembly, inspection, maintenance and testing.

3.3.2 ELECTROMAGNETIC COMPATIBILITY

Electromagnetic compatibility shall be in accordance with the detailed requirements of GE Specification (TBD) , Electromagnetic Compatibility Requirements for Components and Subsystems.

3.3.3 NAMEPLATES AND PRODUCT MARKING

a. Where applicable, components of the TCS shall be marked or tagged for identification in accordance with the manufacturer's standards. The identification shall include, but not be limited to, the following:

1. Nomenclature
2. Customer Part Number
3. Serial Number (Engineering models will use a different designation than prime hardware)
4. Contract Number
5. Manufacturer's Name or Trademark
6. Date of Manufacture (month, day, year)

b. Hardware or equipment which is not suitable for use in flight, and which would be accidentally substituted for Flight or Flight Spares Hardware shall be red striped with material compatible red paint to prevent such substitution. In the event the hardware is too small to be easily striped, or if test results would be affected by striping, a conspicuous red tag marked "NOT FOR FLIGHT USE" shall be attached.

c. Wire and Cables. Wires and cables for hardware shall not be identified by hot stamping directly onto primary or secondary (shield) insulation.

3.3.4 WORKMANSHIP

The components of the TCS, including all parts and subassemblies, shall be constructed, finished and assembled in accordance with the highest standards for high reliability aerospace equipment. Workmanship criteria shall comply with MIL-STD-454, Requirements 9 and 24. Particular attention shall be paid to neatness and thoroughness of soldering, wiring, marking of parts and assemblies, plating, painting, machine screw assemblage and freedom from burrs and shap edges.

3.3.5 CLEANLINESS

Hardware shall be designed, manufactured, assembled, and handled in a manner to insure the highest practical level of cleanliness. Suitable precautions shall be taken to insure freedom from debris within the hardware, and inaccessible areas where debris and foreign material can become lodged, trapped, or hidden shall be avoided. Hardware shall be designed so that malfunctions or inadvertent operating cannot be caused by exposure to conducting or nonconducting debris or foreign material floating in a gravity-free state. Electrical circuitry shall be designed and fabricated to prevent unwanted current paths being produced by such debris. Ultrasonic vibration shall not be used as a method for cleaning component electronic assemblies.

3.3.6 INTERCHANGEABILITY

Each component or part of the qualification vehicle TCS shall be interchangeable with regard to form, fit and function with the flight vehicle components or parts. The requirements of MIL-STD-454, Requirement 7, shall apply.

3.3.7 SAFETY PRECAUTIONS

Warnings and precautions relative to personnel and equipment safety shall be specified in TCS component and part handling, assembly and test instructions.

3.4 DRAWINGS

3.4.1 DRAWING TREE

The TCS drawing tree shall be as defined on Figure 3-2. (Later)

3.5 MAJOR COMPONENTS

3.5.1 INSULATION BLANKETS

3.5.1.1 Description

The blankets shall be composed of multilayers of aluminized films and dacron net for layer separators. The blankets shall be attached to the spacecraft structure and components in appropriate locations and shall be configured to help maintain temperatures and temperature gradients within specified limits.

3.5.1.2 Characteristics

3.5.1.2.1 Thermal Performance

With temperature boundaries of 25°C and -195°C the maximum effective emittance for each blanket type (installed) shall be .015.

3.5.1.2.2 Venting

All blankets shall be vented adequately to limit ballooning to 0.50 inches. during ascent.

3.5.2 THERMAL CONTROL COATINGS

3.5.2.1 Coating Types

3.5.2.1.1 Type A Coatings

Type A thermal coatings shall be used on external vehicle surfaces to reject heat generated by vehicle electrical components while reducing external heat loads. Type A coatings shall have a high hemispherical emissivity and a low solar absorptivity.

3.5.2.1.2 Type B Coatings

Type B thermal coatings shall be used on external vehicle surfaces to reject heat generated by vehicle electrical components without reducing or dampening the orbital variation of external heat loads. Type B coatings have a high hemispherical emmissivity and a high solar absorptivity.

3.5.2.1.3 Type C Coatings

Type C thermal coatings shall be used on external vehicle surfaces to amplify external heat loads while dampening their orbital variation. Type C coatings have a low solar absorptivity and lower hemispherical emissivity with a high α/ϵ ratio.

3.5.2.1.4 Type D Coatings

Type D coatings shall be used on internal vehicle surfaces to maximize heat transfer by radiation. Type D coatings have a very high hemispherical emissivity.

3.5.2.1.5 Type E Coatings

Type E thermal coatings shall be used on internal vehicle surfaces to minimize heat transfer by radiation. Type E coatings have a very low hemispherical emissivity.

3.5.2.2 Locations

The required location for thermal coatings shall be as defined on Table 3-8.

3.5.2.3 Optical Properties

The solar absorptivity at BOL (Beginning of Life) and the hemispherical emissivity of each thermal control coating defined in Table 3-8 shall be measured prior to application.

Table 3-8. Spacecraft Surfaces Optical Property Requirements (TBR)

Location	Solar Absorptivity		Hemispherical Emissivity	Coating Type	Specification
	BOL	EOL			
ACS Module Radiator	0.08	0.17	0.83	Silvered Teflon	171A4632 TY2 M 171A4653 P
Power Module Radiator	0.08	0.17	0.83	Silvered Teflon	171A4632 TY2 M 171A4653 P
C&DH Module Radiator	0.08	0.17	0.83		171A4632 TY2 M 171A4683 P
External Insulation Surface	0.40	TBD	0.76	Aluminized Kapton (Blanket)	S33405 TY5
Antennas	.21	.42	.87	S13G	171A4634
Antenna Support Struts.	.21	.42	.87	S13G	171A4634
Antenna Front	0.21	0.42	0.87	S13G	171A4634
Antenna Back	0.40	TBD	0.76	Aluminized Kapton (Blanket)	S33405 TY5
SAD Shaft	0.90	0.90	N/A	Black Anodize	171A4227 TY1
Shunt Load Panel	.21	.42	.87	S13G	171A4634
Solar Panel Rear Surface	.21	.42	.87	S13G	171A4634
Tank	N/A	N/A	.15 max	Aluminized Kapton Tape	S33069 P3
Valves	N/A	N/A	.15 max	Aluminized Kapton Tape	S33069 P3
Lines	N/A	N/A	.15 max	Aluminized Kapton Tape	S33069 P3
Internal Heat Rejection Surface	TBD	TBD	$\leq$.90	Chemglaze Z306	171A4654
Internal Structure Elements	N/A	N/A	.80 min	Black Anodize	171A4227

3.5.3 THERMAL GREASE

3.5.3.1 Usage

Thermal grease shall be used as defined in Table 3-1 at structure to structure and at component to structure interfaces where mechanical interfaces alone do not permit adequate heat transfer.

3.5.3.2 Conductivity

3.5.3.2.1 Thermal

The grease shall have a thermal conductivity of ≥ 0.73 BTU/hr.ft.°F.

3.5.3.2.2 Electrical

The grease shall have an electrical resistance of ≤ 50 ohms at 33 volts.

3.5.3.3 Evaporation Rate

The grease shall have an evaporation rate of $\leq 0.06\%$ measured for 10 days at 70°F and 10^{-6} mm Hg.

3.5.4 INSULATING WASHERS

3.5.4.1 Usage

The washers shall be used in the component mounting as defined in Table 3-9 to meet critical local thermal conductance requirements not obtainable with the basic spacecraft mechanical interfaces.

3.5.4.2 Conductivity

The thermal conductivity of the basic material shall be 0.12 BTU/hr.ft.°F$\pm$25%.

3.5.5 THERMAL TAPE

3.5.5.1 Usage

Thermal tape shall be used to minimize the heat loss at joints between insulation

Table 3-9. Critical Thermal Conductance Requirements (TBR)

Component Mount	Conductance (Btu/hr.°F)
Mission Independent	
ACS Module	TBD
Power Module	TBD
C&DH Module	TBD
Line Mounts	TBD
Tanks	TBD
Shunt Load Panel	TBD
Latching Valve	TBD
HTE Valve	TBD
Mission Peculiar	
Wideband Module	TBD
Thematic Mapper Module	TBD
MSS	TBD
TDRSS	TBD

Table 3-10. Heater Assemblies (TBD)

Location	Number	Max. Pwr(ω)	Thermostat Temp. Control Ranges
ACS Module	10	13.6	66 $\pm$ .5°F
C&DH Module	10	10.0	66 $\pm$ .5°F
Power Module	10	11.1	46 $\pm$ .5°F
Thrust Valves	14	11.8	40 $\pm$ .5°F
Catalyist Bed	14	22.4	Command Activated

blanket assemblies, or where local substrate requires the optical surface properties of the external tape surface.

3.5.5.2 Optical Properties

The solar absorptivity at BOL and hemispherical emissivity shall be measured for compliance with the requirements of Table 3-8 prior to application.

3.5.5.3 Adhesion

The thermal tape must maintain its required optical properties after being installed and exposed to the maximum substrate temperature profile.

3.5.6 HEATERS

The heaters shall be foil-type bonded in place.

3.5.6.1 Usage

The heater assembly shall be utilized only where thermal control cannot be accomplished by passive means and where a maximum temperature control band range of TBD is permissable.

3.5.6.2 Requirements

The heater assembly power capability shall be as defined in Table 3-10 for the nominal thermostat set point.

3.5.6.3 Size

The heaters shall be physically sized by controlling surface areas and thickness, for each user location, to provide uniform heat input. Hot spots shall be eliminated.

3.5.6.4 Electrical Resistance

The electrical resistance of each heater shall be measured at maximum nominal and minimum system voltage.

3.5.6.5 Redundance

There shall be redundant heaters at each location given in Table 3-10.

3.5.6.6 Thermostat

3.5.6.6.1 Type

The thermostat shall be electrically activated, per General Electric Drawing (TBD) .

3.5.6.6.2 Redundancy

Each heater assembly shall have two thermostats, one with the nominal set point, and one in series with a set point below the minimum "on" point for the nominal set point.

3.5.6.6.3 Characteristics

The closing point temperature and opening point temperature of each thermostat shall be measured.

3.5.6.6.4 Selection

In establishing the circuit defined in paragraph 3.5.6.6.2, the thermostat characteristics obtained in Section 3.5.6.6.3 shall be used as follows:

a. The total activation temperature range for each thermostat shall be less than 1°F(TBR).

b. The minimum deadband between the thermostats shall be 1°F(TBR).

c. The minimum closing point of the low temperature power on thermostat shall be equal to or greater than the minimum temperature requirement of the component being heated.

SECTION 4

VERIFICATION

4.1 GENERAL

Performance verification of the TCS shall be accomplished to demonstrate compliance with the specifications in the paragraphs under Section 3.

4.2 VERIFICATION METHODS

Table 4.1, Requirements Verification Matrix, defines the method of verification for each requirement specified in Section 3. Verification methods, and requirements for their use, are defined in subparagraphs of paragraph 4.2.

4.2.1 SIMILARITY

Verification by similarity shall be used if it can be shown that the article is substantially similar or identical in design, manufacturing processes and quality control to another article that has been previously qualified to equivalent or more stringent criteria.

4.2.2 ANALYSIS

Analytical techniques may be used, in lieu of testing to verify compliance to specified requirements, either along or in combination with test. The selected techniques may include, typically, system engineering analysis, statistics, qualitative analysis, analog modeling, and computer simulation.

4.2.3 INSPECTION

Inspection may be used to verify design and construction requirements, drawing compliance, or specific physical dimensions of the item.

4.2.4 TESTS

No verification testing of the TCS will be performed at the subsystem level. Verification testing shall be performed at the spacecraft level only.

Table 4-1. Requirements Verification Matrix

Section 3 Paragraph	N/A	Verification Method: Similarity 4.3	Inspection 4.4	Analysis 4.5	4.6 Test: Qual 4.6.1	Accept 4.6.2
3.1 Thermal Control Subsystem Definition	X					
3.1.1 Thermal Control Subsystem Description			X	X		
3.1.1.1 TCS Block Diagram				X		
3.1.2 Interface Definition	X					
3.1.2.1 External Interfaces	X					
3.1.2.1.1 Fairing			X	X		
3.1.2.1.1.1 Surface Emissivity			X			
3.1.2.1.1.2 Fairing Blanket				X		
3.1.2.1.2 Launch Vehicle			X	X		
3.1.2.1.3 Spacecraft AGE			X	X		
3.1.2.2 Internal Interfaces	X					
3.1.2.2.1 TCS Component Interfaces			X			
3.1.2.2.2 ACS Module	X					
3.1.2.2.2.1 ACS Component Interface			X			
3.1.2.2.2.2 ACS Component Thermal Dissipation				X		
3.1.2.2.2.3 Antenna Feed Assembly			X			
3.1.2.2.3 C&DH Module	X					
3.1.2.2.3.1 C&DH Component Interface			X			
3.1.2.2.3.2 C&DH Component Thermal Dissipation				X		

Table 4-1. Requirements Verification Matrix

Section 3 Paragraph	N/A	Verification Method: Similarity 4.3	Inspection 4.4	Analysis 4.5	4.6 Test: Qual 4.6.1	Accept 4.6.2
3.1.2.2.4 Power Module	X					
3.1.2.2.4.1 TCS Component Power Source			X			
3.1.2.2.4.2 Harness Interface			X			
3.1.2.2.4.3 Electrical Short			X			
3.1.2.2.4.4 Command Capability			X			
3.1.2.2.4.5 Power Component Interface			X			
3.1.2.2.4.6 Power Component Thermal Dissipation				X		
3.1.2.2.4.7 Thermostat Electrical Interface			X	X		
3.1.2.2.4.8 Flight Temperature Sensors			X			
3.1.2.2.5 Propulsion Module			X			
3.1.2.2.6 Structure Subsystem	X					
3.1.2.2.6.1 Ground Conditioning			X			
3.1.2.2.6.2 TCS Installation			X			
3.1.2.2.6.3 Flight Sensor Locations			X			
3.1.2.2.6.3.1 Sensor Bonding				X		
3.1.2.2.6.4 Ascent Flight	X					
3.1.2.2.6.4.1 Time-Temp. Profile-Fairing Acoustic Blanket				X		

Table 4-1. Requirements Verification Matrix

Section 3 Paragraph	N/A	Verification Method: Similarity 4.3	Inspection 4.4	Analysis 4.5	4.6 Test: Qual 4.6.1	Accept 4.6.2
3.1.2.2.7 Mission Peculiar Modules	x					
3.1.2.2.7.1 Heat Exchange				X		
3.1.2.2.7.2 Cryogenic Coolers			X			
3.2 Characteristics	X					
3.2.1 Performance	X					
3.2.1.1 General				X		
3.2.1.2 Temperature				X		
3.2.1.3 Thermal Dissipations				X		
3.2.1.4 Mission Phases				X		
3.2.1.5 Attitude Control				X		
3.2.1.6 Useful Life				X		
3.2.2 Physical Characteristics	X					
3.2.2.1 Installation			X			
3.2.2.2 Size			X			
3.2.2.3 Weight			X			
3.2.2.4 Power				X		
3.2.2.5 Quantity			X			
3.2.2.6 Transportation			X			
3.2.3 Reliability				X		
3.2.4 Maintainability			X			
3.2.5 Environmental Conditions	X					

Table 4-1. Requirements Verification Matrix

Section 3 Paragraph	N/A	Similarity 4.3	Inspection 4.4	Analysis 4.5	4.6 Test: Qual 4.6.1	4.6 Test: Accept 4.6.2
3.2.6 Transportability				X		
3.3 Design and Construction	X					
3.3.1 Materials, Processes and Parts	X					
3.3.1.1 Selection of Materials, Processes and Parts			X			
3.3.1.2 Selection of Electronic Parts			X			
3.3.1.3 Screening of Parts			X			
3.3.1.4 Parts Specifications			X			
3.3.1.5 Part Application Restrictions			X			
3.3.1.6 Parts Derating			X			
3.3.1.7 Traceability of Parts			X			
3.3.1.8 Corosion Prevention			X			
3.3.1.9 Moisture and Fungus Resistance			X			
3.3.2 Electromagnetic Compatability				X		
3.3.3 Nameplates and Product Marking			X			
3.3.4 Workmanship			X			
3.3.5 Cleanliness			X			
3.3.6 Interchangeability			X			
3.3.7 Safety Precautions			X			
3.4 Drawings	X					
3.4.1 Drawing Tree			X			
3.5 Major Components	X					

Table 4-1. Requirements Verification Matrix

Section 3 Paragraph	N/A	Verification Method: Similarity 4.3	Inspection 4.4	Analysis 4.5	4.6 Test: Qual 4.6.1	Accept 4.6.2
3.5.1 Insulation Blankets	X					
3.5.1.1 Description			X			
3.5.1.2 Characteristics	X					
3.5.1.2.1 Thermal Performance				X		
3.5.1.2.2 Venting				X		
3.5.2 Thermal Control Coatings	X					
3.5.2.1 Coatings Types	X					
3.5.2.1.1 Type A Coatings				X		
3.5.2.1.2 Type B Coatings				X		
3.5.2.1.3 Type C Coatings				X		
3.5.2.1.4 Type D Coatings				X		
3.5.2.1.5 Type E Coatings				X		
3.5.2.2 Locations			X			
3.5.2.3 Optical Properties				X		
3.5.3 Thermal Grease	X					
3.5.3.1 Usage			X			
3.5.3.2 Conductivity	X					
3.5.3.2.1 Thermal				X		
3.5.3.2.2 Electrical				X		
3.5.3.3 Evaporation Rate				X		
3.5.4 Insulating Washers	X					

Table 4-1. Requirements Verification Matrix

Section 3 Paragraph	N/A	4.3	4.4	4.5	4.6 Test: Qual 4.6.1	4.6 Test: Accept 4.6.2
3.5.4.1 Usage			X			
3.5.4.2 Conductivity				X		
3.5.4.3 Installation				X		
3.5.5 Thermal Tape	X					
3.5.5.1 Usage			X			
3.5.5.2 Optical Properties				X		
3.5.5.3 Adhesion				X		
3.5.6 Heaters	X					
3.5.6.1 Usage			X			
3.5.6.2 Requirements				X		
3.5.6.3 Size			X			
3.5.6.4 Electrical Resistance			X	X		
3.5.6.5 Redundance			X			
3.5.6.6 Thermostat	X					
3.5.6.6.1 Type			X			
3.5.6.6.2 Redundancy			X			
3.5.6.6.3 Characteristics				X		
3.5.6.6.4 Selection			X	X		

SECTION 5.0

SPECIFICATION

FOR THE

EOS COMMUNICATIONS & DATA HANDLING SUBSYSTEM MODULE

TABLE OF CONTENTS

SECTION 1

SCOPE

This specification establishes the performance, design, interface, and verification requirements for the C&DH subsystem to be used on the EOS-A spacecraft and other earth orbiting spacecraft. For adaptability to the command, telemetry, clock, timecode, and data handling requirements of different missions, the C&DH subsystem will be modularized, with the addition or deletion of equipments (OBC memory modules, CPU, NBTR, TDRSS transponder) dependent on the mission requirements. In addition, command, telemetry, and timecode distribution will be done with data busses which service varying numbers of users without impacting the C&DH interface.

SECTION 2

APPLICABLE DOCUMENTS

The following documents of the exact issue shown, form a part of this specification to the extent specified herein. In the event of conflict between documents referenced here and the detail requirements in the following sections, the detail requirements of this specification shall supersede. In the event of conflict between documents referenced here and lower tier references to documents referenced here, the former supersede.

2.1 APPLICABLE DOCUMENTS

SPECIFICATIONS

National Aeronautics and Space Administration

EOS-410-02	Specifications for EOS System Definition Studies, 13 September 1974
S-311-P-11	Quality Monitoring of Integrated Circuit, 1 June 1970
S-323-P-10	Connectors, Subminiature Electrical and Coaxial Contacts for Space Flight Use, Revised December 1969

MILITARY

MIL-C-38999	Connectors, Electrical, Miniature, Quick Disconnect, Est. Reliability
MIL-C-29012A	Connectors, Coaxial, RF, General Specification for
MIL-C-26482	Connectors, Electric, Circular, Miniature, Quick Disconnect
MIL-C-17	Cables, RF, Coaxial, Dual Coaxial, Twin Conductors, Twin Head
MIL-W-81044	Wire, Electric Cross-linked, Polyalkene, Insulated, Copper
MIL-E-5400K	Electronic Equipment, Airborne, General Specification for

General Electric

SVS XXXX	Specification for EOS General Purpose Spacecraft Segment
SVS XXXX	Specification for EOS Basic Software

STANDARDS

National Aeronautics & Space Administration

Aerospace Data Systems Standard, Part II, Section 3, PCM Command Data System Standard, prepared by GSFC Data Systems Requirements Committee, 23 April 1968

Aerospace Data System Standards, Part III, Associated System STandards, Section 2, "Spacecraft Clock Systems Standard", prepared by GSFC Data Systems Requirements Committee, April 1964

Aerospace Data System Standards, Part III, Associated Standards, Section I, "Radio Frequency and Modulation Standard for Space-to-Ground Telemetry", prepared by GSFC Data Systems Requirements Committee, November 1965

Aerospace Data Systems Standard, Part I, Telemetry Standards, Section 1,"Pulse Code Modulation Telemetry Standard", prepared by GSFC Data Systems Requirements Committee, January 27, 1966

Part III, Associated Systems Standards, Section 3, "Spacecraft Minitrack Signal Source Standard", prepared by GSFC Data Systems Requirements Committee, October 1963

Military

MS33540C	Safety Wiring, General Practices for
MIL-STD-454B	Standard General Requirements for Electronic Equipment
MIL-STD-143A Change 1	Specification and Standards, Order of Precedence for Selection of
MS-33586A	Metal, Definition of Dissimilar
MIL-STD-130C	Identification Marking of U.S. Military Property

OTHER PUBLICATIONS

National Aeronautics and Space Administration

NHB 5300.4 (3A) May 1968	Requirements for Soldered Electrical Connections
PPL-12 Latest Issue	GSFC Preferred Parts List
NHB 5300.4 (1A)	Reliability Program Provision for Space Systems Contractors
NHB 5300.4 (1B)	Quality Assurance Program Provisions for Space Systems Contractors

Military Handbooks

MIL-HDBK-5A	Metallic Materials and Elements for Aerospace Vehicle Structure
MIL-HDBK-17	Plastics for Flight Vehicles

General Electric

XXXXX	EOS General Purpose Spacecraft Quality Program Plan
XXXXX	Configuration Management Plan for EOS General Purpose Spacecraft
XXXXX	Reliability Program Plan, EOS General Purpose Spacecraft

SECTION 3

REQUIREMENTS

3.1 ITEM DEFINITION

3.1.1 ITEM DESCRIPTION

The C&DH subsystem provides spacecraft tracking, ground and on board control of all spacecraft and payload sensor functions, and retrieval of narrowband and mediumband ($\leq$650 kHz) observatory data. A block diagram of the subsystem is shown in Figure 3.1.

The antenna is an omni-directional slotted cylinder used for both transmit and receive at S-band. The STDN transponder is a phase lock loop S-band transponder whose downlink carrier frequency is determined by a coherent sample of the received uplink carrier (when present). Received data are demodulated from the PM'd carrier and provided as an output to the modulation processor. Input data from the modulation processor PM the downlink carrier. The modulation processor uses a Costas loop demodulator to obtain command data from the STDN 70 kHz subcarrier or obtains demodulated command data from the TDRSS xpdr. These command data are used to develop corresponding bit sync and enable signals and all three signals are output to the central command decoder. The modulation processor also selects, conditions, and convolutionally encodes (TDRSS only) data and provides composite output signals as modulation inputs to the STDN & TDRSS transponders. The central command decoder decodes uplink command data from STDN or TDRSS and determines its destination (OBC for stored command or memory load; TFG for application to the supervisory data bus).

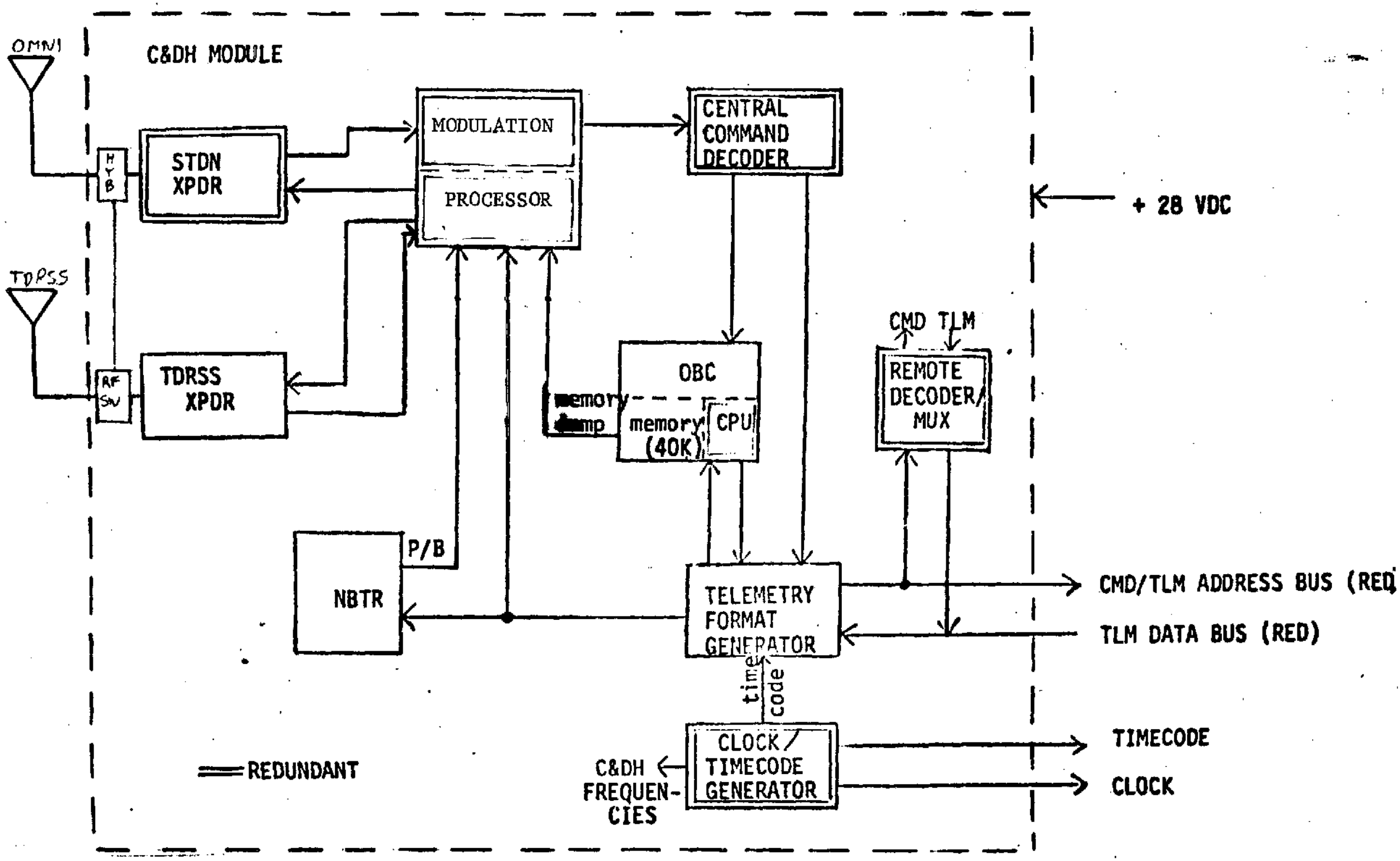


Figure 3.1. C&DH Block Diagram

It loads the output data into a register and provides a ready signal to the OBC or TFG for subsequent readout. The telemetry format generator controls the operation of the supervisory and return busses which provide access to the remote decoder/muxes. It controls the sequencing of data on the supervisory bus by periodically sampling its internal ROM (or RAM) and the output registers of the CCD and OBC. It also controls the formatting, data rate, and time buffering of telemetry data to be transmitted to the ground. Major frame, minor frame, and word rate signals are developed and inserted into the telemetry address words on the supervisory data bus. The remote decoder/muxes accept command data from the supervisory data bus and feed telemetry data to the return data bus. They act as the command and telemetry interface with each of the module components. Each provides 64 pulse command outputs, four serial magnitude outputs, and 64 telemetry inputs (analog, bilevel digital, and up to 16 serial digital). The on-board computer is an AOP with five 8K core memory modules. It performs on-board computational, analytical, and control functions by issuing commands to and requesting data from the remote decoder/muxes located in each of the spacecraft subsystems. It interfaces with the CCD, TFG, and umbilical with DMA I/O's. The clock and timecode generator uses a 3.2 MHz temperature compensated oscillator for deriving all clock frequencies needed by components within the C&DH module. It also provides separately buffered 1.6 MHz balanced outputs for use as a standard clock by other spacecraft subsystems. It generates a 32 bit, 1 msec timecode for use by the TFG and spacecraft subsystems for data annotation. The NBTR is the NASA/GSFC 10^9 recorder and provides back orbit data storage for analysis of critical events and failure modes. The TDRSS transponder is an S-band transponder which demodulates the uplink PSK PN code data, extracts ID and command data (if any), and applies a correlated, locally generated PN code to the downlink for ranging. It also accepts convolutionally encoded data from the modulation processor for PSK of the downlink. Uplink command data are demodulated and provided as an input to the modulation processor for generation of bit sync and enable signals for the CCD.

All of the C&DH components except the antenna are housed in a 16 x 40 x 48 inch module. The antenna is located on a 40 inch boom at the forward end of the spacecraft.

3.1.2 INTERFACE DEFINITION

3.1.2.1 Electrical

The electrical interface with the C&DH module is accomplished through six separate connectors mounted on the surface of the module. Two coaxial RF connectors are provided for interface with the omni-directional antenna and the S-band feed of the TDRSS antenna. A third connector provides the power, command, telemetry, clock, timecode and medium band data ($\leq$650 kHz) signals. The signal distribution on this connector is given in Table 3-1. The fourth connector provides an interface with the spacecraft umbilical connector consisting of two OBC DMA I/O channels (used to decode and format shuttle data) and a hardwire connection to the central command decoder input and the telemetry format generator output for use in the absence of RF links. The fifth connector provides a test interface for checkout of module functions using GSE. A sixth (RF) connector provides an interface between a DCS antenna and a DCS within the module (if present).

3.1.2.2 Mechanical

The C&DH module is bolt mounted directly to the spacecraft structure. The antenna is mounted to a 40 inch boom on the instrument support structure.

Table 3-1. Electrical I/F

Signal	# Pins	Cable
Reg Bus Input	2+2 RTN	T2
Heater Power Input	1+1 RTN	T2
Supervisory Data Bus Output	2+2 RTN	T2S
Return Data Bus Input	2+2 RTN	T2S
1.6 MHz Clock Output	6+6 RTN	Twin-ax
Timecode Output	2+2 RTN	SCS
Medium band Data Input	1	SC
Module Signal Ground	2	SC
Shield Tie (Chassis Gnd)	10	--

3.1.2.3 Thermal

The C&DH module is thermally isolated from the spacecraft structure. Heat generated within the module is rejected from the outboard surface of the module. Thermal control is provided by heaters serviced by the heater bus and controlled by the OBC.

3.2 CHARACTERISTICS

3.2.1 PERFORMANCE

3.2.1.1 RF

The C&DH module shall contain a transponder(s) which is capable of communicating at S-band with both STDN and TDRSS. This transponder shall be used for ranging, commanding, and transmission of narrowband and mediumband (<650 kHz) sensor and housekeeping data.

STDN communications shall be compatible with the NASA/GSFC Aerospace Data System Standards X-560-63-2 and have the following characteristics:

Transmit Frequency	2200 to 2300 MHz $\pm$0.001%
Receive Frequency	2050 to 2150 MHz
Turn around Ratio	221/240
Ranging Sidetone Frequency	500 kHz maximum
Command Bit Rate	2000 bps
Command Modulation	PSK/PM (70 kHz subcarrier)
Narrowband Data Rate	Selectable: 16 kbps, 8 kbps, 4 kbps, 2 kbps 1 kbps
Narrowband Modulation	Bi-ϕ PCM/PM on 1250 kHz subcarrier
Mediumband Data Rate	650 kHz maximum
Mediumband Modulation	Bi-ϕ PCM/PM on carrier
NB and MB Data Coding	Manchester
Analog Data Bandwidth	$\pm$ 250 kHz
Analog Data Modulation	FM frequency translated to 2.25 MHz
Transmitter Power	1 watt, minimum
Modulation Bandwidth	$\pm$2.5 MHz

The downlink shall be capable of simultaneously transmitting the narrowband data; the mediumband data; and analog sensor data @500 kHz bandwidth. Mediumband data shall be selectable by command and consist of GRARR return data, OBC memory dump @128 kbps, NBTR playback @20X narrowband rate, or sensor data $\leq$650 kHz. Data shall be received/transmitted via an omni-directional antenna located external to the C&DH module and having the following characteristics:

Type	Slotted Cylinder
Gain (free space)	-6 dBI, minimum
Gain (120° earth viewing cone)	-1 dBI, minimum
Spacecraft Blockage	15% maximum

TDRSS communications shall be compatible with the NASA/GSFC TDRSS Users Guide, X-805-74-176 and have the following characteristics:

Transmit Frequency	2200 - 2300 MHz (SA), 2287.5 MHz (MA)
Receive Frequency	2025 - 2120 MHz (SA), 2106.4 MHz (MA)
Ranging	PN code $\oplus$ ID
Command Bit Rate	100-1000 bps
Forward Link Modulation	PSK
Narrowband Data Rate	Selectable: 16 kbps, 8 kbps, 4 kbps, 2 kbps, 1 kbps
Narrowband Modulation	PSK on carrier
Mediumband Data Rate	560 kbps, maximum
Mediumband Modulation	PSK on carrier
NB & MB Data Coding	
Transmitter Power	2 watts, minimum

A unique PN code in phase with the received code shall be generated for the return link and half-added to the convolutionally encoded narrowband telemetry data. Ranging and narrowband data shall be selectable by command with medium-band data (OBC memory dump, NBTR playback, or sensor data). Data shall be received by the omni antenna defined above or by the S-band feed of the 8 foot TDRSS dish (defined in wideband C&DH specification). Data shall be transmitted via the eight foot dish.

3.2.1.2 Command and Telemetry

The C&DH module shall provide the decoding and distribution of command data and the acquisition and formatting of narrowband telemetry data. These data shall be distributed and collected via remote decoder/muxes located within each of the spacecraft subsystems. Communications between the C&DH module and the remote units shall be via two data busses: supervisory and return.

These data busses shall also provide the communication link between the on-board computer (OBC) and the spacecraft subsystems. The supervisory data bus shall be capable of handling up to 50 commands per second from the ground along with

16 kwords of telemetry address data and 15.95 kwords of OBC command data. The return data bus shall be capable of handling up to 32 kwords per second of response data.

Uplink command data will be received at a rate between 100 bps (TDRSS) and 2000 bps (STDN) and will be compatible with the NASA/GSFC PCM command standard. Each command message will begin with a 40 bit synchronization word consisting of 39 logical 0's followed by a logical 1 and continue with 40 bit command words having the format shown in Figure 3-2. The first seven bits identify the spacecraft address (unique for each spacecraft). The next two bits are an operations code which identifies the type of command data to be processed (00=realtime, 01=computer data, 10=delayed command data, 10=delayed time tag). The next twenty four bits contain the command data. The last seven bits are a polynomial check code on the entire command word. Invalid commands shall be flagged in telemetry and, if realtime, rejected.

An operations code of 00 or 10 identifies data to be eventually applied to the supervisory data bus and, hence, requires further breakdown of the 24 bit command data. The first five bits identify one of 32 remote decoder/muxes tied to the supervisory data bus. The next bit identifies the data as a pulse command or a serial magnitude command. The next two bits identify which of four serial magnitude outputs of a given remote decoder/mux is to be activated. The final 16 bits contain the serial magnitude command data or identify (last 6 bits) one of 64 remote/decoder mux outputs for execution of a pulse command. The central command decoder shall distribute these data to either the OBC or the telemetry format generator depending on the operations code.

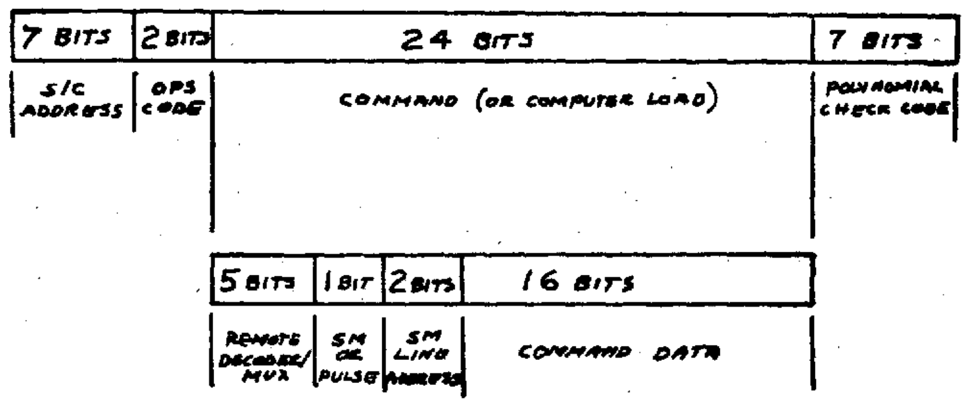


Figure 3-2. Uplink Command Word Format

Data on the supervisory data bus shall be time division multiplexed from three sources: (1) a RAM (or ROM) within the TFG which controls the generation of a telemetry matrix by specifying the remote decoder/mux address and gate ID to obtain data for transmission on the downlink; (2) the OBC which requests data from the remote decoder/muxes for computational purposes or issues commands to the remote decoder/mux as a result of computation, spacecraft status, or delayed command timetag; (3) the CCD which formats realtime commands from the ground. The TFG shall format these data into 32-bit words (Manchester encoded) for transmission to the remote decoder/muxes. The formatting is shown in Figure 3-3. The sync bits shall be illegal Manchester bits to identify the beginning of a word. The address bits shall identify the remote decoder/mux (one of 32) and the command type bits identify which logic to energize. The power strobe bits shall be unused bit times (3 usec) to permit power switching transients in the selected remote to dissipate. (In the case of a telemetry address word these bits shall also be used to identify the first word of each major and minor frame and matrix word rate). The remainder of the word shall be used to identify the output/input for performance of the selected function and for a parity check of the word.

Data transmitted on the supervisory data bus shall be Manchester encoded and allocated to pre-determined time slots. These time slots shall be 31.25 usec. wide, permitting 32000 32 bit words per second to be transmitted at a 1.024 Mbps bus rate. Every 640th slot shall be allocated for use by the CCD for executing realtime commands, which may be received at a maximum rate of one each 20 msec. At a downlink telemetry rate of 16 kbps, every sixteenth slot shall be allocated to the RAM (or ROM) addresses. (This allocation shall be proportional to the downlink telemetry rates of 16, 8, 4, 2 or 1 kbps). All other slots on the data bus shall be allocated to use by the OBC as necessary.

The C&DH module output shall be transformer coupled and TTL compatible.

Each remote decoder/mux, when interrogated, shall apply a data word in serial form to the return data bus. The data word shall be a Manchester encoded 16 bit word containing status, parity, and data as shown in Figure 3-4. The status bits shall contain data indicating the response of the remote (TBD), the parity check bit shall indicate the result of the supervisory word parity check, the 8 bit data word is only used for telemetry data, the parity bit is on the return word. The return data word shall be transmitted at 1.024 Mbps during the first 16 bit times following the supervisory data word. The C&DH module input shall be transformer coupled and TTL compatible.

The supervisory and return data busses shall be capable of supporting up to 32 remote decoder/muxes. These remotes shall obtain regulated bus input power from the user subsystem. Each remote shall be transformer coupled to the supervisory data bus and contain TTL compatible circuitry. It shall contain sentry logic which checks each word on the supervisory data bus for its address. If its address is recognized it shall turn on power to the remainder of its logic needed to perform the function. Each remote shall have 64 pulse command outputs and four serial magnitude outputs. Characteristics of these outputs are given below:

<u>Pulse Commands</u>

Pulse Duration	20 ms
Logical "1"	+5V @ 4 ma
Logical "0"	0 to +.5V
R Source @ "0"	8.0K ohms maximum

<u>Magnitude Commands</u>

Clock Rate	1.024 Mbps
Gate Width	Envelopes 16 clock pulses
Command Word	16 bits serial

These signal outputs have the same voltage and impedance characteristics as those shown for pulse commands

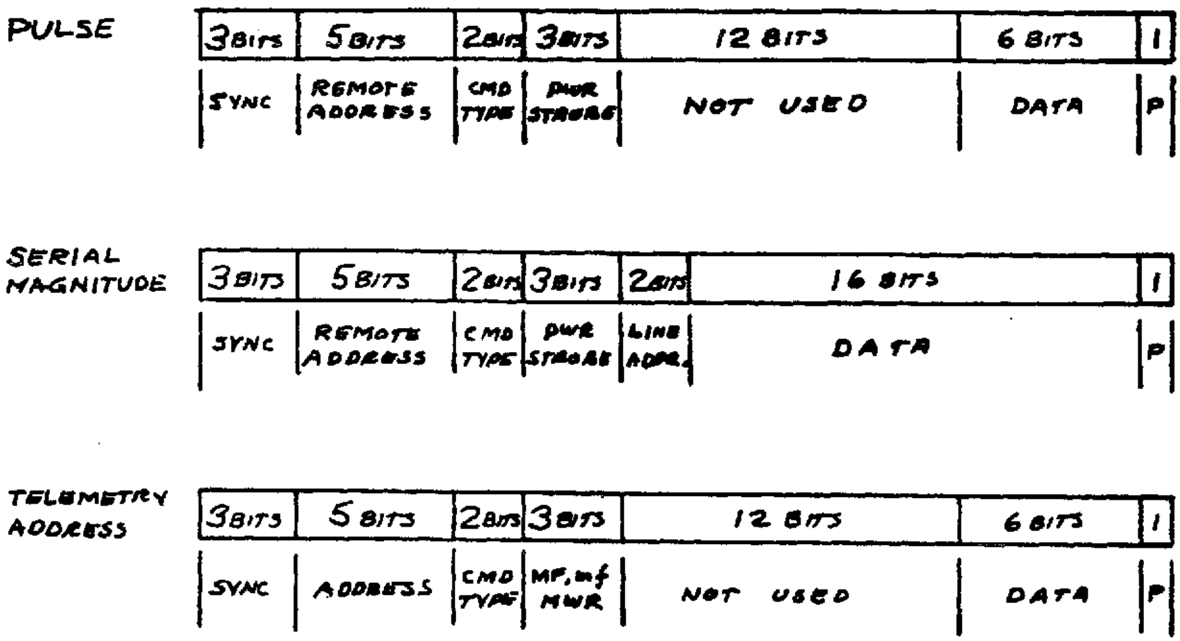


Figure 3-3. Supervisory Data Bus Word Format

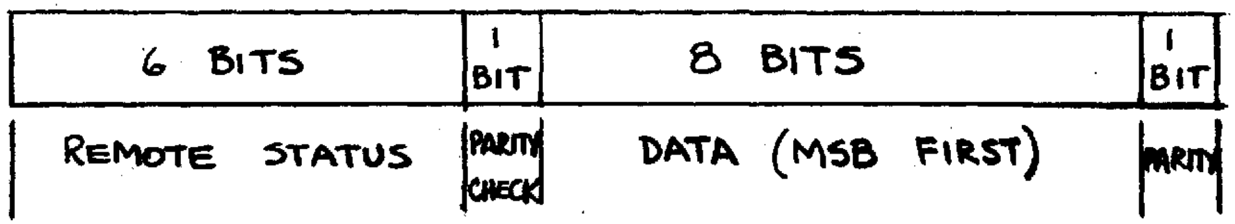


Figure 3-4. Return Data Bus Format

Each remote shall also have 64 inputs that can be used for analog, bilevel digital, and serial digital signals. Each input shall be capable of handling any type of data, except that serial digital signals shall be limited to 16 per remote. These inputs shall accept signals with the following characteristics:

Analog Inputs (Digitized to 8 bits)

Range	0 to +5 VDC
Z Source	5K ohms maximum
Accuracy	+40 MV

Bilevel Digital Inputs

Logical "1"	+5 VDC
Logical "0"	0 $\pm$ 0.5 VDC
Fault Tolerance	-20 to +40 VDC
Z Source	5K ohms minimum; 10K ohms maximum

Serial Digital Inputs (8 bits/word)

Input Data	
Logical "1"	+5 VDC
Logical "0"	0 $\pm$ 0.5 VDC
Z Source	5K ohms maximum
Output Data	
Clock Rate	1.024 mbps
Gate Width	Envelopes 8 clock pulses

These outputs shall have the same voltage and impedance characteristics as those shown for magnitude commands.

All inputs of the remote shall have an input impedance of 10 megohms minimum during non-sampling periods and 10K ohms minimum during sampling. The inputs shall be capable of surviving a short circuit to +28 VDC. The remotes shall also provide major frame, minor frame, and word rate output signals to the user subsystem. These signals shall have the same impedence and voltage characteristics as pulse commands.

All data on the return data bus shall be fed to the OBC for analysis and/or computation. Data obtained in response to a RAM (or ROM) address shall be formated into 128 x n word frame (n is binary number $\leq$ 128) consisting of 124 columns of sampled data (including 4 subcommutated) and transmitted to the ground. The format shall be controlled by the ROM (replaceable prior to launch) or the RAM (reprogrammable through the OBC). Either shall be selectable by command. Synchronization, minor frame ID, and timecode data shall be inserted into each major frame of data. Data rate to the ground shall be selectable by command as 1,2,4,8, or 16 kbps. Major frame,minor frame, and word rate shall be provided to the remotes as part of the telemetry address words on the supervisory data bus.

3.2.1.3 On-Band Computer

A general purpose digital computer shall be included in the C&DH subsystem. The computer shall communicate with all spacecraft subsystems and devices through time shared use of the supervisory and return data busses.

40K words of non-volatile memory shall be provided in 8K word segments. Cycle-by-cycle power switching of the memory shall be employed. Implementation of the memory shall be such that the function of any bank, including that used for fixed locations, can be achieved by any other bank.

All input and output data channels shall be designed to operate in both a direct memory access (DMA) mode and program control mode. One application of the DMA shall be the loading and dumping of any set of memory locations independent of processor operation and of memory content. The dump format shall include memory address and content.

The Central Processor Unit (CPU) design shall include, the following.

1. One index register
2. 16 bit word size minimum.
3. 16 maskable interrupts minimum.
4. Add instruction (<5 microseconds).
5. Multiply instruction.

6. Divide instruction.
7. Four basic logical instructions minimum (AND, OR, EX OR, and COMPLEMENT).
8. Condition/unconditional transfer.
9. Protection against illegal write to memory.

Even though memory protect (Item 9) is a hardware function, capability shall exist for modifying via interrupt the selection of memory segments to be protected. During program execution, write cycles will be prohibited in the instruction portion of memory. To be consistent with this restriction, a "transfer and set return instruction" shall not require a write cycle in the protected area.

The OBC shall have the following characteristics:

Add	5 usec
Multiply	38 usec
Divide	75 usec
Word Length	18 bits, minimum
Memory Type	Core
Memory Access Time	500 usec
Memory Cycle Time	1.2 usec
DMA Time	10 usec
DMA Channels	10
I/O Execute Time	10 usec
Interrupt Levels	16

3.2.1.4 <u>Narrowband Tape Recorder</u>

The C&DH module shall be capable of housing a 10^9 bit digital narrowband tape recorder which will be used to record the narrowband telemetry data when the spacecraft is not in contact with a ground station. Input and output data shall be Manchester encoded with the playback rate 20 times the record rate.

3.2.1.4 <u>Frequency & Timecode Generator</u>

The C&DH module shall contain a frequency and timecode generator for providing a standard clock reference and timecode annotation for use by all the spacecraft subsystems. The frequency generator shall use a 3.2 MHz oscillator as a stable reference (1 x 10^8 per year) and derive all frequencies needed by components within the C&DH module. It shall also provide separately buffered 1.6 MHz balanced output drivers for use as a standard clock by other S/C subsystems. These outputs shall have a source impedance of 78 ohms $\pm$10% and an output voltage of 2v p.p The timecode generator shall provide a 32 bit Manchester encoded timecode representing milliseconds of a month. This timecode shall be inserted into the four subcommutated columns in the first minor frame of each major frame. It shall also be available as a 35 khz data bus to other S/C subsystems. The first 3 bits of each 35 bit word shall be illegal Manchester data as on the supervisory data bus. This data bus shall be transformer coupled to all users and provide a TTL compatible output.

3.2.2 DESIGN

3.2.2.1 Electrical

3.2.2.1.1 Power

The C&DH module shall be capable of operating from a regulated bus input voltage of +28 $\pm$0.3 VDC containing a voltage ripple $\leq$ 100 millivolts p-p. The source impedance of the bus will not exceed 0.1 ohms from 0 to 10 khz. The C&DH module shall be capable of surviving power system failure modes which can vary the bus voltage from +18 to +33 volts with a volt second product $\leq$ 250 uvolt - seconds. The C&DH module shall not present a positive or negative step load changes greater than 28 $\pm$ 2 volts with a volt second product $\leq$ 100 uvolt seconds. The rate of current rise or fall shall not exceed TBD amperes per second. Power input shall be provided on two T2 cables. The orbital average power dissipation of the C&DH module shall not exceed 182 watts.

3.2.2.1.2 Command

Command control of the components in the C&DH module shall be obtained via two remote decoder/muxes located within the module. The decoded commands will be output as logic drive signals as defined in Para. 3.2.1.2. Relay driver circuits shall be provided within the components as necessary. A List of the C&DH command requirements is provided in Table 3-2.

3.2.2.1.3 Telemetry

Telemetry data acquisition from each of the C&DH module components shall be obtained via two remote decoder/muxes located within the module. The telemetry inputs shall be conditioned to meet the requirements of Para. 3.2.1.2. A list of the C&DH telemetry data requirements is provided in Table 3-3.

3.2.2.1.4 Outputs

The C&DH module shall provide three RF coaxial interfaces. One shall interface with the omni-directional S-Band antenna; one shall interface with the S-Band feed of the TDRSS antenna; one shall be available as an interface with a turnstile antenna on the external surface of the C&DH module and an internal DCS. The supervisory and return data busses and the timecode data bus shall be provided as redundant T2S cable outputs with one of each bus energized at any one time (selectable by command). The 1.6 MHz clock outputs shall be six separately buffered twin-ax cables. An interface shall also be provided with the spacecraft umbilical connector which provides access with two OBC DMA channels and with the CCD input and the TFG output.

3.2.2.1.5 Grounding

Three separate grounding busses shall be established within the C&DH module. A power ground bus shall tie the primary return of all component input power DC/DC converters together at a common point within the module which, in turn, shall be tied to the input power return lines from the power module. This return shall

C-3

Table 3-2. Command Requirements

(TBD)

Table 3-3. Telemetry Data Requirements

(TBD)

also be used as the return for all relay driver current. A signal ground bus shall be established within the module and shall be used to tie all component signal grounds together and, in turn, to the spacecraft unipoint ground reference via two single conductor cables. This ground shall be used as a reference for all signals not tied to power or chassis ground (telemetry, logic command, time code, 1.6 MHz clock, DC/DC convertor secondary). The third ground is module chassis which is electrically tied to the spacecraft structure. All module components housings shall be electrically tied to the module frame. This ground is used as a signal reference for all RF devices and as a shield tie point.

All three data bus returns (supervisory, return, and timecode) shall be floating.

3.2.2.1.6 Redundancy

Redundancy within the C&DH module shall be limited to those components necessary to track and command the S/C in order to permit retrieval by shuttle. These include the S-band transponder, command demodulator and modulator linear summer in the modulation processor, the CCD, the data bus drivers in the TFG, the data busses and remote decoder/muxes, and the frequency and timecode generator. The CPU and power switching networks of the OBC shall also be redundant to provide increased reliability in the performance of critical spacecraft functions.

3.2.2.1.7 Electromagnetic Compatibility

The C&DH subsystem module shall be designed to minimize the radiation of self-generated noise and shall be well shielded to preclude the possibility of susceptibility to EMI from spacecraft or external sources. System design shall be based on the suppression of noise at its source and the containment of self-generated noise within the generating assembly. Good design practices in chassis design, EMC filtering, grounding, bonding, etc. shall be employed throughout the program.

3.2.2.1.8 Harness

The module harness shall provide all electrical interfaces between subsystem assemblies within the module and to the module/structure interface and test connectors. The harness shall be of modular design for maximum system flexibility. Installation or removal of the harness shall be possible without removing electrical assemblies. It shall be possible to remove electrical assemblies without removing the harness. Cable strain relief or backshell potting shall be employed at all harness terminations.

Wire sizes shall be selected to hold round trip voltage drops between source and load to one percent or less of the supply voltage. The minimum wire size for power and control circuitry shall be AWG #20. The minimum wire size for data or test circuitry shall be AWG #22. Under worst case conditions, wire temperature shall not exceed the temperature rating of the wire insulation.

3.2.2.2 Mechanical

The C&DH module equipment shall be housed in a container having dimensions of 16 x 40 x 48 inches. Mechanical characteristics including dimensions, mounting flange locations, handling lug size and location, etc. shall conform to the standard spacecraft module specification (TBD).

The internal assemblies of the C&DH module shall have simple bolt mounting and electrical connector interfaces. These assemblies shall be mounted to the module to meet required thermal and environmental conditions.

The weight of the C&DH Module shall not exceed 240 pounds.

SPECIFICATION SVS-XXXX
16 SEPTEMBER 1974

SECTION 6.0

SPECIFICATION

FOR THE

EARTH OBSERVATORY SATELLITE (EOS)

BASIC SOFTWARE

TABLE OF CONTENTS

SECTION 1

SCOPE

This specification establishes the performance and interface requirements for the application packages necessary for operation of the basic spacecraft bus for EOS. Software application packages are included for the command, telemetry, and power functions. Each of these packages defines the interfaces and computational requirements associated with the function and includes a flow diagram showing the basic approach to the software package. Requirements for the ACS applications package are included as part of the ACS Specification and are not included in this document.

SECTION 2

APPLICABLE DOCUMENTS

TBD

SECTION 3.0

REQUIREMENTS

3.1 COMMAND STORAGE AND SEQUENCING APPLICATION PACKAGE

The Command Storage and Sequencing software application package (CSS) shall store the following in OBC memory: (1) delayed commands and associated time tags received from the ground via the CCD and (2) those commands, generated by other application packages due to the expiration of a time interval, the occurrence of a special vent or alarm (as determined from data in the TLM stream), or the result of a computational routine. The CSS shall periodically examine these stored commands, placing those ready for issuance in an output buffer, ready for handling by the Executive.

Four buffers shall be used by the CSS: one for a storage of delayed commands and associated time tags (received from the Central Command Decoder) which have been received in chronological order; a second for those commands and time tags (also received from the CCD) which are not chronologically ordered; a third for the designation of those commands and command sequences specified by other application packages; and a fourth for storage of all the commands and command sequences which can be specified by the other OBC software.

Upon institution of the CSS by the Executive, the CSS shall perform the following functions: (1) compare the time tag associated with the first command in the ordered buffer with the contents of the Real Time Clock (CRTC); (2) compare the time tags associated with each of the commands in the unordered buffer with the contents of the RTC; (3) place in an output command buffer ready for issuance by the Executive those commands whose time tags match the contents of the RTC; (4) assemble in the output command buffer those commands and command sequences which are specified in the command designator buffer; (5) modify the pointer(s) associated with the ordered and/or unordered buffers if a time tag comparison(s) has been achieved in either or both of these buffers, and (6) transfer control to the Executive with a control word designating whether a command(s) is to be issued. See the attached flow chart of the CSS operation, Figure 3.1-1.

3.2 TELEMETRY DATA HANDLING APPLICATION PACKAGE

The Telemetry Data Handling software application package (TDH) shall perform the following functions:

(1) Storage of a telemetry format matrix comprising addresses relating to 128 minor frames (rows) of 128 words each. Each minor frame is identical except for four subcommutated words. Each address consists of 13 bits - (5 bits for the remote decoder/mux address, 6 bits to specify the specific data source to be sampled, and 2 bits to designate the type of data to be transferred). This matrix is transferred viz a DMA channel to the TFG RAM when a command to do so is received by the TDH.

(2) Generation of addresses for transmission (via the TFG) to the different subsystems and payload to obtain special data for the purpose of performing various checks.

(3) Performance of status, limits, and alarm checking on the telemetry (TLM) data and the special data received (via the TFG) in response to the addresses referred to in (2).

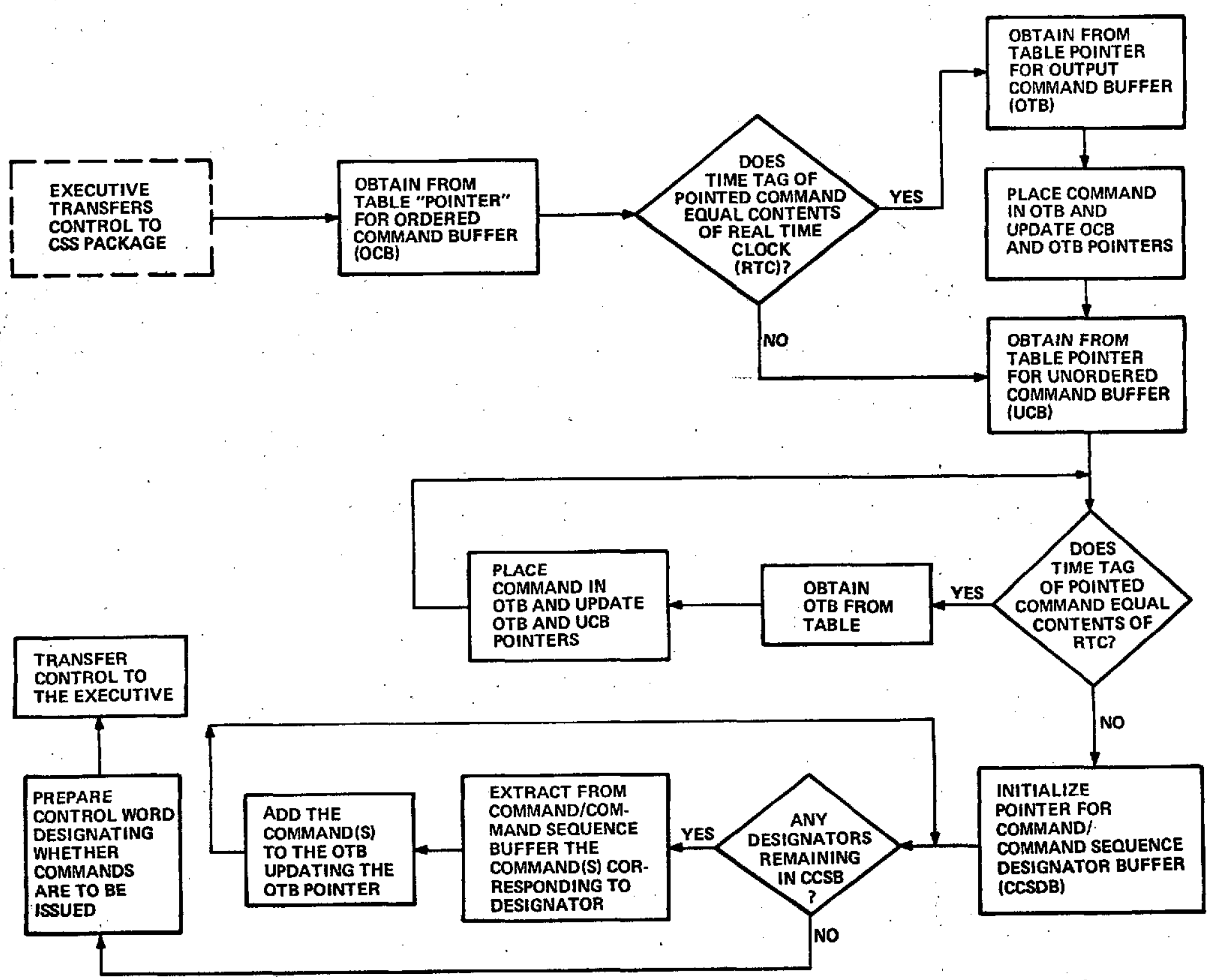


Figure 3.1-1. Command Storage and Sequencing Application Package (CSS)

(4) Generation of command and command sequences as a function of the results of the checks performed on the TLM and the special data.

(5) Injection into the TLM data stream, for transmission on the downstream link, data reflecting the results of the checks performed.

The initiation of the transferrence of the telemetry format matrix shall be precipitated by a command from the ground which is referred to the TDH. The TDH then generates control words which identify a command or command sequence which shall have been pre-stored in the command/command sequence buffer. These control words, which shall also designate the memory area in which the telemetry format matrix is stored, will be stored in the command/command sequence designator buffer. The subsequent CSS processing of these control words will lend to the Executive initiating the DMA transfer of the matrix to the TFG.

The generation of special addresses to the TFG to obtain subsystem/payload data for checking purposes shall be performed at preestablished intervals in accordance with information prestored in the TDH special address table. Status checking is performed on key parameters relating to the operational status of each spacecraft subsystem. A maximum of 200 status parameters are monitored. A record is maintained in memory of status changes of significance - e.g., those which were not anticipated. This record is transferred to the ground during an OBC memory dump. In addition, the TDH may initiate the issuance of a command/command sequence.

Limit checking is performed for up to 100 selected analog telemetry parameters each of which is checked against an upper and lower limit to insure subsystem performance safety. Out of limit conditions result in either a telemetry "flag" inserted into the TLM stream, a memory record established for subsequent memory dump to the ground, and/or initiation of a command/command sequence.

Alarm checks monitor up to 25 specific S/C events. Occurrence of an alarm condition will result in either inspection into the TLM stream of an appropriate flag which will subsequently cause an OBC memory dump to be requested, initiation of a response program to be performed by the OBC, or activation of a command/command sequence with appropriate notification to the ground. (See the attached flow chart of the TDH operation, Figure 3.2-1.)

3.3 POWER MANAGEMENT APPLICATION PACKAGE

The Power Management application package (PM) shall perform the following functions:

(1) Monitor the voltage and current at each of up to 10 loads.

(2) Accumulate and monitor load power consumption.

(3) Monitor the charge/discharge rate and the temperature of each of up to 5 batteries.

(4) Determine the operating point control for each battery.

(5) Control by command the charge/discharge rate of each battery so as to conform to operating point control.

(6) When appropriate, issue commands to perform load switching and deactivation of a battery(ies).

The PM shall perform the load voltage and current monitoring function by periodically examining the values of the appropriate load sensors located at the output of the power module. The sensor values are obtained from the TLM stream, and are transferred into the OBC memory from the TFG by the Executive

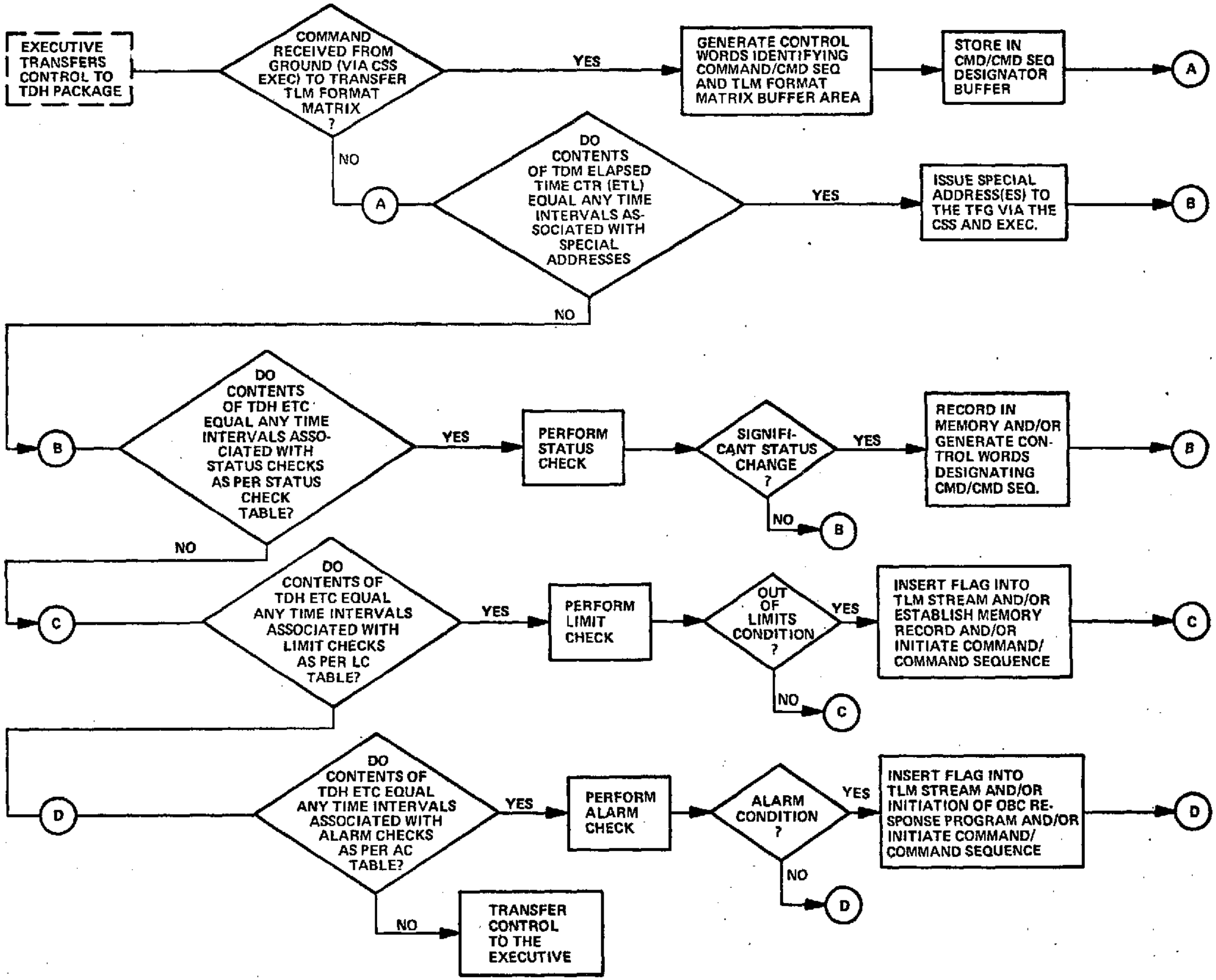


Figure 3.2-1. Flow Chart of Telemtry Data Handling Application Package (TDH)

using the DMA technique. When the PM detects an out of limits condition(s), the PM will issue commands to (a) effect load switching (deletion), and (b) input into the downlink TLM stream on indication of the detected condition.

The PM shall also use its load current monitoring function to accumulate on an orbital basis the total amp/minutes used by each load. The PM will check for an abnormal deviation in load demand during any given orbit. If such an abnormal change is detected, it is considered to be an alarm condition and will be treated by the PM in a manner similar to that employed by the Telemetry Data Handling Application Package.

The PM shall determine the battery operating point control for each of up to five batteries by assessing the scheduled operating time of the S/C subsystems and payload, the charge/discharge profile of the battery during the previous orbit, the present battery temperature, and the charge/discharge status of the battery (expressed in the accumulation of amp/minutes charged and discharged) during the present orbit. The AP will generate charge/discharge commands (via the Command Storage and Sequencing Application Package) to each battery to conform to the battery operating point control. The AP will command high rate charging until a full charge is established; subsequent overcharging is performed at a reduced current level to avoid pressure buildup.

The PM shall monitor the battery temperature of each battery. If this temperature does not fall within prescribed limits, the PM shall issue commands (to be executed via the CSS) to deactivate the faulty battery and to inject into the downlink TLM streams an indication of this alarm condition. A flow chart for the Management Application Package is shown in Figure 3.3-1.

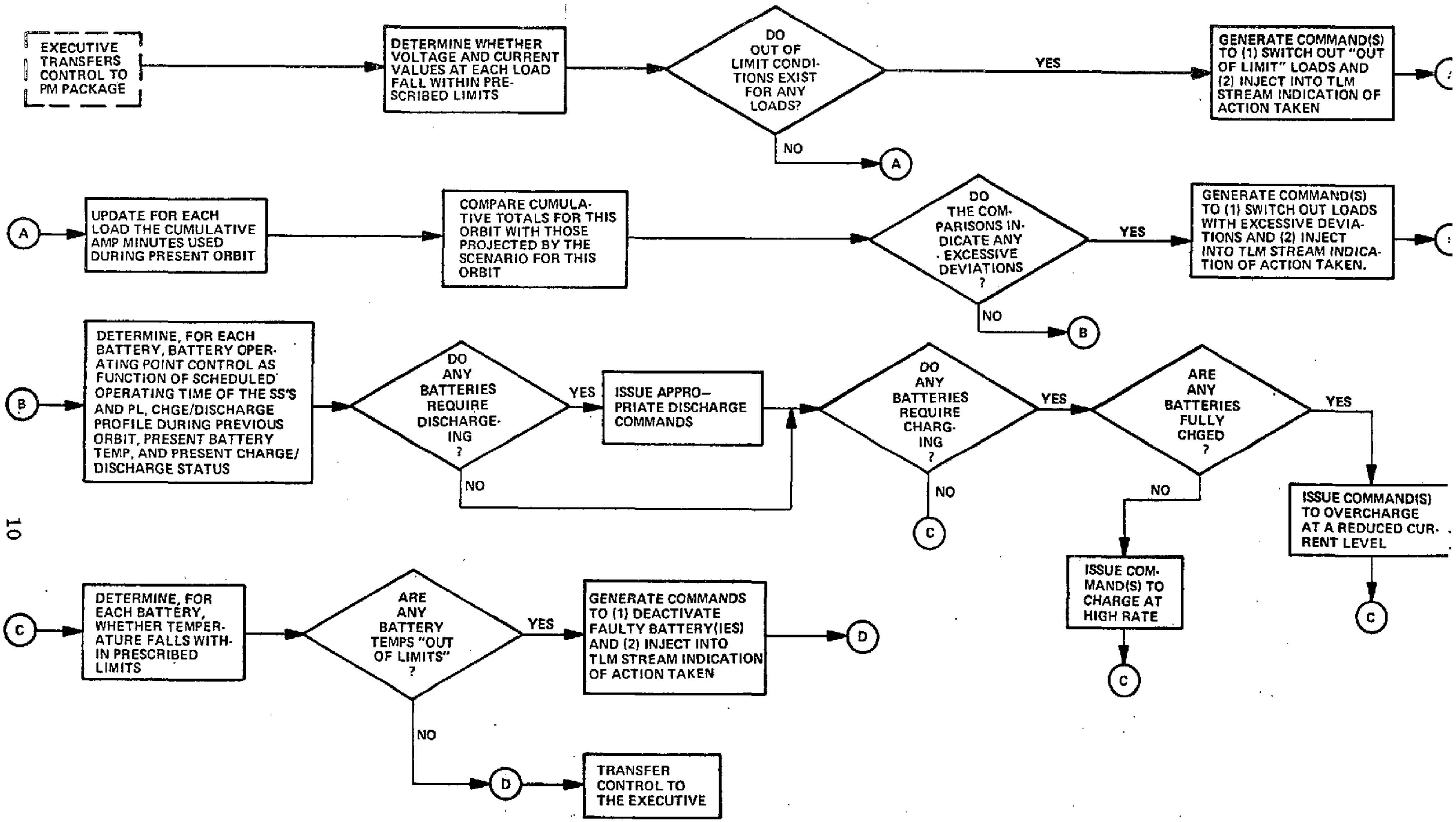


Figure 3.3-1. Flow Chart of Power Management Application Package

SECTION 7.0

SPECIFICATION FOR THE

EARTH OBSERVATORY SATELLITE
ATTITUDE CONTROL SUBSYSTEM MODULE

TABLE OF CONTENTS

SECTION 1.0

SCOPE

This specification establishes the requirements for performance, design, qualification and acceptance testing of a subsystem identified as the ATTITUDE CONTROL SUBSYSTEM, hereinafter referred to as the subsystem. The subsystem is used to provide accurate control of spacecraft angular attitude required by the Earth Observatory Satellite. The subsystem requires earth, sun, and star references for pointing purposes.

SECTION 2.0

APPLICABLE DOCUMENTS

The following documents of the exact issue shown, form a part of this specification to the extent specified herein. In the event of conflict between documents referenced here and the detail requirements in the following sections, the detail requirements of this specification shall supersede. In the event of conflict between documents referenced here and lower tier references to documents referenced here, the former supersede.

SPECIFICATIONS

National Aeronautics and Space Administration

EOS-410-02	Specifications for EOS System Definition Studies, 13 Sept. 1974
S-311-P-11	Quality Monitoring of Integrated Circuits, 1 June 1970
S-323-P-10	Connectors, Subminiature Electrical and Coaxial Contacts for Space Flight Use, Revised December 1969

MILITARY

MIL-C-38999	Connectors, Electrical, Miniature, Quick Disconnect, Est. Reliability
MIL-C-29012A	Connectors, Coaxial, RF, General Specification for
MIL-C-26482	Connectors, Electric, Circular, Miniature, Quick Disconnect
MIL-C-17	Cables, RF, Coaxial, Dual Coaxial, Twin Conductors, Twin Head
MIL-W-81044	Wire, Electric Cross linked, Polyalkene, Insulated, Copper
MIL-E-5400K	Electronic Equipment, Airborne, General Specification for

GENERAL ELECTRIC

SVS XXXX	Specification for EOS General Purpose Spacecraft Segment
SVS XXXX	Specification for EOS Reaction Control Propulsion Subsystem Module

MILITARY

MS35540C	Safety Wiring, General Practices for
MIL-STD-454B	Standard General Requirements for Electronic Equipment

MIL-STD-143A Change 1	Specification and Standards, Order of Precedence for selection of
MS-33586A	Metal, Definition of Dissimilar
MIL-STD-130C	Identification Marking of US Military Property
MIL-STD-1247A	Identification of Pipe, Hose, and Tube Lines for Aircraft, Missile and Space Systems

OTHER PUBLICATIONS

National Aeronautics and Space Administration

NHB 5300.4 (3A) May 1968	Requirements for Soldered Electrical Connections
PPL-12 Latest Issue	GSFC Preferred Parts List - June 1970
NHB 5300.4 (1A)	Reliability Program Provisions for Space Systems Contractors
NHB 5300.4 (1B)	Quality Assurance Program Provisions for Space Systems Contractors

MILITARY HANDBOOKS

MIL-HDBK-5A	Metallic Materials and Elements for Aerospace Vehicle Structure
MIL-HDBK-17	Plastics for Flight Vehicles

GENERAL ELECTRIC COMPANY

XXXXX	EOS General Purpose Spacecraft Quality Program Plan
XXXX	Configuration Management Plan for EOS General Purpose Spacecraft
XXXX	Reliability Program Plan, EOS General Purpose

SECTION 3.0

REQUIREMENTS

3.1 ITEM DEFINITION

3.1.1 ITEM DESCRIPTION

This subsystem provides control for each of three spacecraft axes: roll, pitch and yaw. It provides the capability for acquisition of references, for pointing with primary sensors, and for pointing with alternate sensors. The subsystem provides a primary pointing mode by orienting the spacecraft with respect to a periodically updated inertial reference frame. Spacecraft orientation in the earth oriented mode is defined with respect to the reference frame indicated in Figure 3-1 with positive yaw along the local vertical, positive roll nominally along the orbit velocity vector and perpendicular to Yaw and positive pitch along the orbit normal such that a right handed coordinate frame is formed. The subsystem also provides attitude control during orbit maintenance and orbit adjust maneuvers through use of reaction control jet modules. The normal mode of operation is achieved using momentum wheel torques with unloading from magnetic torquers through the earth's magnetic field. A back-up wheel unload mode provides control using the reaction jet thrusters for orbit maintenance, and a back-up pointing mode provides control through use of the acquisition sun sensors. This subsystem is configured as indicated in Figure 3-2.

3.1.2 Interface Definition

3.1.2.1 Schematic Arrangement

See Figure 3-3.

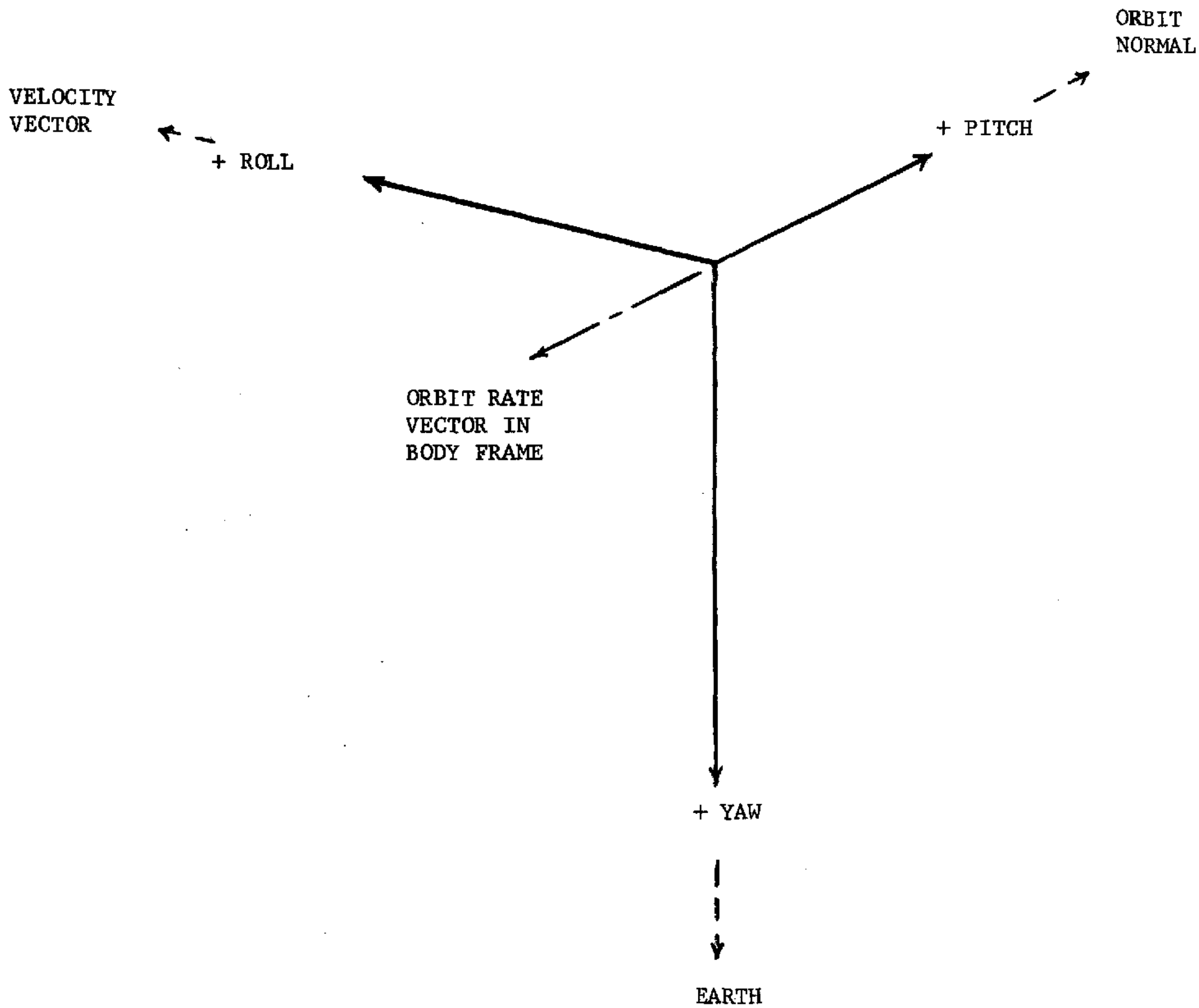


Figure 3-1. Axis Definition

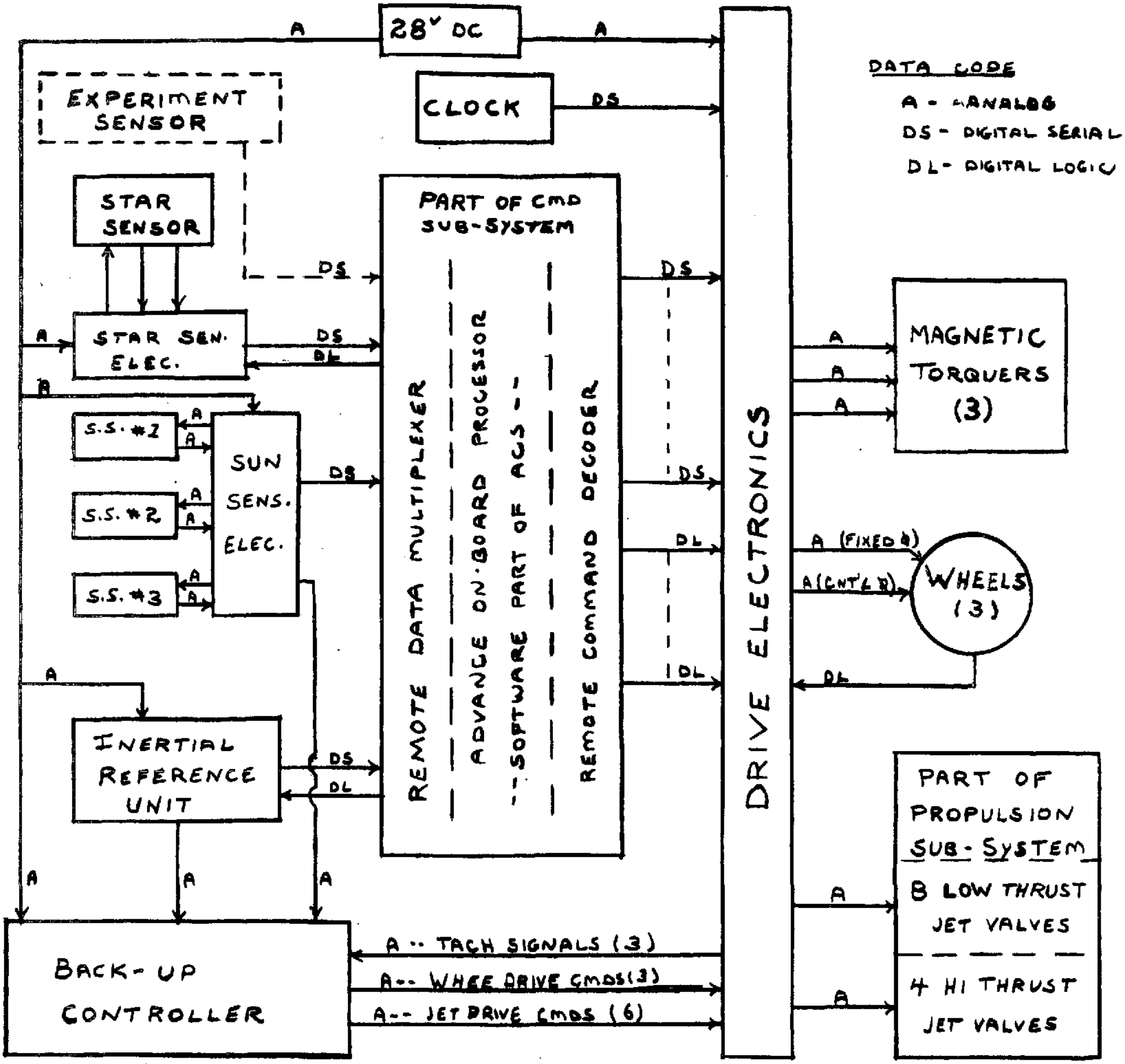


Figure 3-2. Functional Block Diagram

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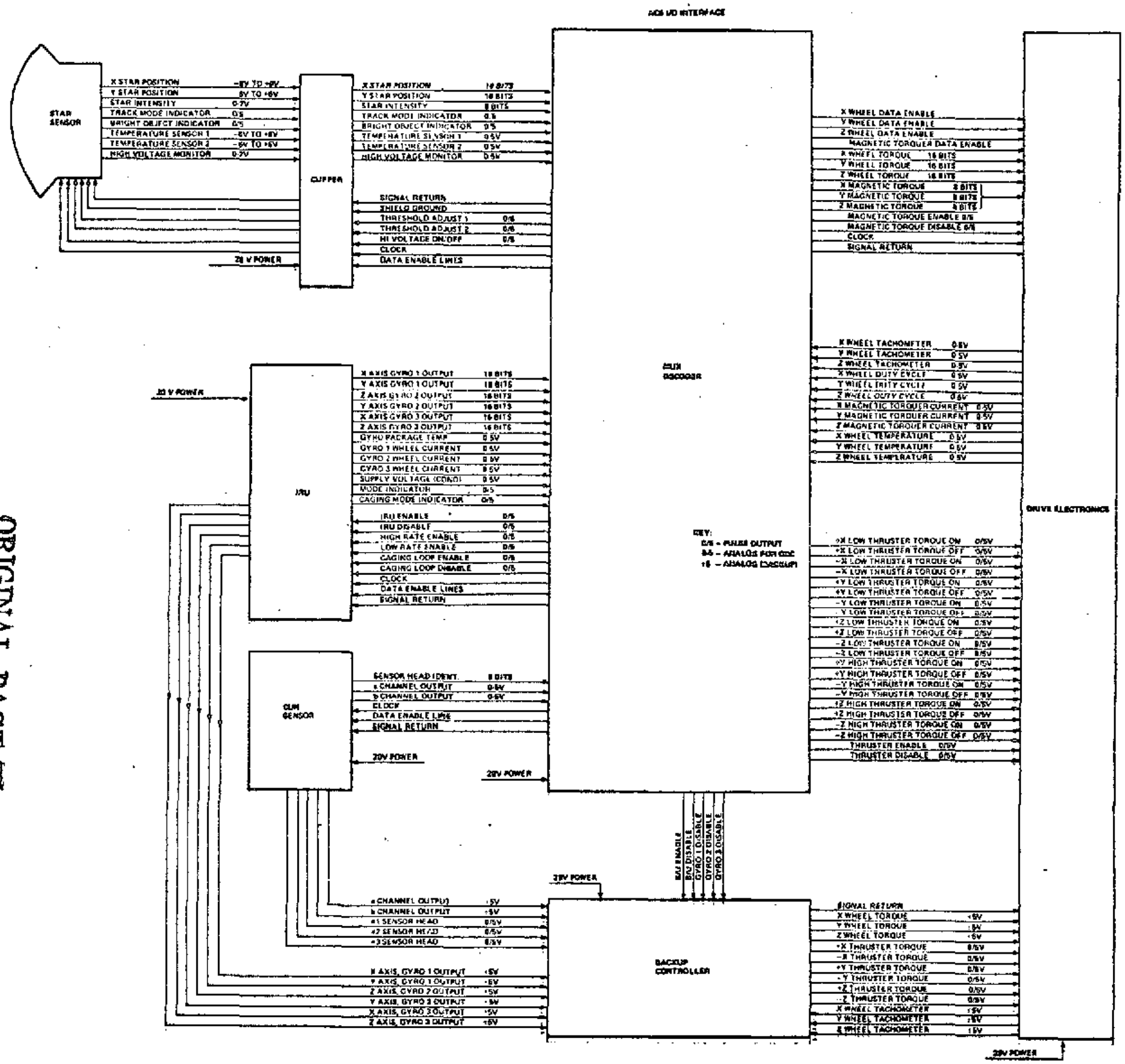


Figure 3-3. Schematic Arrangement

3.1.2.2 Subsystem Module Structure

All ACS components requiring alignment shall be referenced to the star sensor during subsystem module testing per paragraph 3.2.2.4 except for two of the three digital sun sensor heads which are not mounted in the ACS module.

The ACS module will mount to the subsystem structure such that the star sensor reference frame will be aligned to the vehicle reference within 6-arc minutes. The two digital sun sensor heads not located in the ACS module will be aligned to the vehicle reference to within 0.5 degrees .

3.1.2.3 Command & Data Handling Subsystem

This subsystem shall provide for the following interfaces with the command subsystem

a) Mechanical: The ACS module shall supply space for the location of a standard remote command decoder and a standard remote multiplexer for the purpose of sending digital and analog data to the on-board computer and receiving digital command data from the on-board computer.

b) Electrical: Signals to be supplied to the on-board computer will be analog, digital, or pulse, as shown in Figure 3-3. Analog data shall have the following characteristics:

   o linear range: zero to five volts
   o maximum saturation: five volts
   o source impedance: 5000 ohms
   o reference level: analog reference to be supplied by the user subsystem, referenced to subsystem signal ground.

The digital data sent to the remote multiplexers shall be serial, NRZ data, shifted in 8 bit bytes according to a series of 8 shift pulses. A coded interrogate signal identifying the word required by the multiplexer will precede the shift pulses on a separate line.

Signals supplied to this subsystem through the command decoder will be parallel, NRZ, 8 bit words or discrete levels per Figure 3-3. The subsystem shall transfer the parallel data to sample and hold digital-to-analog converters for processing according to their function. All discrete data shall be held in electronic latching circuits by the subsystem until an independent complementing discrete is received to reset the latch.

3.1.2.4 Power Subsystem

This subsystem shall be capable of operating within the requirements of this specification when provided with regulated input power having the following characteristics:

1. Operating Voltage: 28.0 volts dc, $\pm$ .3v dc

2. Noise and Ripple: 100 millivolts peak to peak

3. Transients: 100 microvolt-second

4. Switching Requirements: The turn-on/off operation of this subsystem of its components shall not cause a current ramp whose slope exceeds 100,000 ampheres per second.

5. Isolation and Grounding: TBD

6. Average Input Power: 50 watts orbital average power

7. Maximum Input Power: 425 watts

8. Abnormal Condition: This subsystem shall be capable of surviving, without damage, an instantaneous voltage between +18 and +33 volts dc, with a disturbance product no greater than 250 microvolt seconds.

3.1.2.5 Propulsion Subsystem

This subsystem shall provide for the following interfaces with the Propulsion Subsystem:

a) Electrical: Each control valve (12) of the Propulsion Subsystem shall be supplied power via two lines from the drive electronics. One line shall carry regulated dc voltage as indicated below, the other shall be a return. The return line shall be connected to a switch which shall be either open circuit or shall connect the return line to the ground return to the power supply. The output of the drive electronics shall be measured across the two lines to the valve. With the switch in the open state, the output shall be less than 1.0 volt dc; with the switch in the closed state, the output shall be not less than 24.0 volts dc for all valves. The valve winding will have a resistance of 200 ohms $\pm$ 10%, in series with an inductance of less than 70 millihenrys.

The requirements stated above shall be met over all required conditions of temperature, life, or regulated power supply voltages as indicated.

b) Acceleration: Each low thrust control jet will provide spacecraft acceleration equal to or less than 1.46×10^{-3} rad/sec.^{2} The minimum pulse increment of the low thrust jets will not exceed .01 pound-sec.

Each medium thrust control jet will provide spacecraft acceleration equal to or less than 6.25×10^{-3} rad/sec.^{2} The minimum pulse increment of the high thrust jets will not exceed .6 #-sec.

3.1.2.6 AGE Interfaces

This subsystem shall provide for the following interfaces with AGE:

a) Electrical

b) Mechanical: Per drawing (TBD)

c) Alignment: Per drawing (TBD)

3.2 CHARACTERISTICS

3.2.1 PERFORMANCE

3.2.1.1 Primary Subsystem Performance Characteristics

In this specification the pointing requirements cited are for the subsystem alone. In the case of operation with the primary sensors, the sensor error exclusive of its mounting error is considered part of the subsystem error. In the case of operation with experiment sensors (which are not part of this subsystem), the requirements cited are with respect to the sensor output signal as received by this subsystem. For initial acquisition the sensors are the sun sensor, the star tracker, and the inertial reference unit.

3.2.1.1.1 Sun Line Acquisition

The Subsystem shall be capable of pointing both null axes of the negative yaw sun sensor elements to the sun to an accuracy of 0 $\pm$1 degrees (3 sigma) and reducing the spacecraft yaw rate to 0 $\pm$.05 degree/second starting from any spacecraft attitude, and with maximum yaw, roll, and pitch rates of 1.0 degree/second. During non-eclipsed periods, the sun line shall be acquired within 30 minutes from receipt of external command when rates are less than 1.0 degrees/second about each axis and from any initial attitude.

3.2.1.1.2 Control to Sun Line

The subsystem shall be capable of offset bias attitude control about either or both of the negative yaw sun sensor sensitive axes to within ± 1 degree of the bias commands. Total angle of the bias command (cone angle from the sun line) will not exceed 20 degrees.

Rate control about the sun lines shall be 0 $\pm .05$ degrees/second.

3.2.1.1.3 Celestial Reference Acquisition

The star sensor shall respond to a preselected star (or star pattern) in the presence of a spacecraft rate command about the sun line of 0.2 $\pm .05$ degrees/second. Upon receipt of this star recognition response, the subsystem shall set a software quaternion reference, based on star sensor data, while continuing to rotate about the specified spacecraft axis. Addditional reference updates as known stars pass the star sensor field of view shall be processed to improve three axis position accuracy without use of sun sensor data. The initial star recognition sequence shall be complete within 60 minutes of receipt of the search rate command and three axis reference accuracy to 1 arc-minute (exclusive of the contributions of gyro error sources) shall be obtained within 30 minutes after the first reference update.

3.2.1.1.4 Operational Control

The subsystem shall be capable of implementing the three independent mission profiles described in subsequent paragraphs.

a. Earth oriented in which a negative spacecraft pitch axis rate is commanded such that the positive spacecraft yaw axis is directed toward the earth's centroid and the spacecraft roll axis is directed along the velocity vector. Accuracy in controlling the spacecraft to this reference frame shall be within 36 arc-sec (1 sigma) per axis after 5 orbits of reference update data on stars of opportunity. Position jitter as a function of frequency at the end of this interval shall be within the frequency profile of Figure 3-4 (3 sigma).

b. Inertial hold in which all spacecraft rates shall be reduced to 0 $\pm$.003 degrees per-hour after sufficient reference update time, in a low rate reference acquisition mode, is allowed to calibrate gyro drift (100 minutes). Without the use of attitude sensors and without on-orbit calibration, inertial hold rates shall be within $\pm$.5 degree/hour (3 sigma). Position accuracy with respect to an arbitrary inertial reference shall be 36 arc-sec (1 sigma) after calibration.

c. Slew mode in which the subsystem shall be capable of executing a 90 $\pm$.03 degree eigenvector rotation profile with rates up to .2 degrees per minute. The use of available stars for reference updates during this maneuver is assumed.

3.2.1.1.5 Momentum Management

The subsystem shall be capable of maintaining less than 30 percent momentum wheel speed in normal mode for low orbit altitudes (but greater than 300 nm) without the use of reaction control jets. For synchronous altitude missions, reaction jets shall be used for wheel unloading. During the unloading maneuver, the spacecraft attitude accuracy shall be within 0.1 degree of the nominal. Settling to normal mode performance shall be accomplished within 60 seconds after receipt of a single axis wheel unload sequence command.

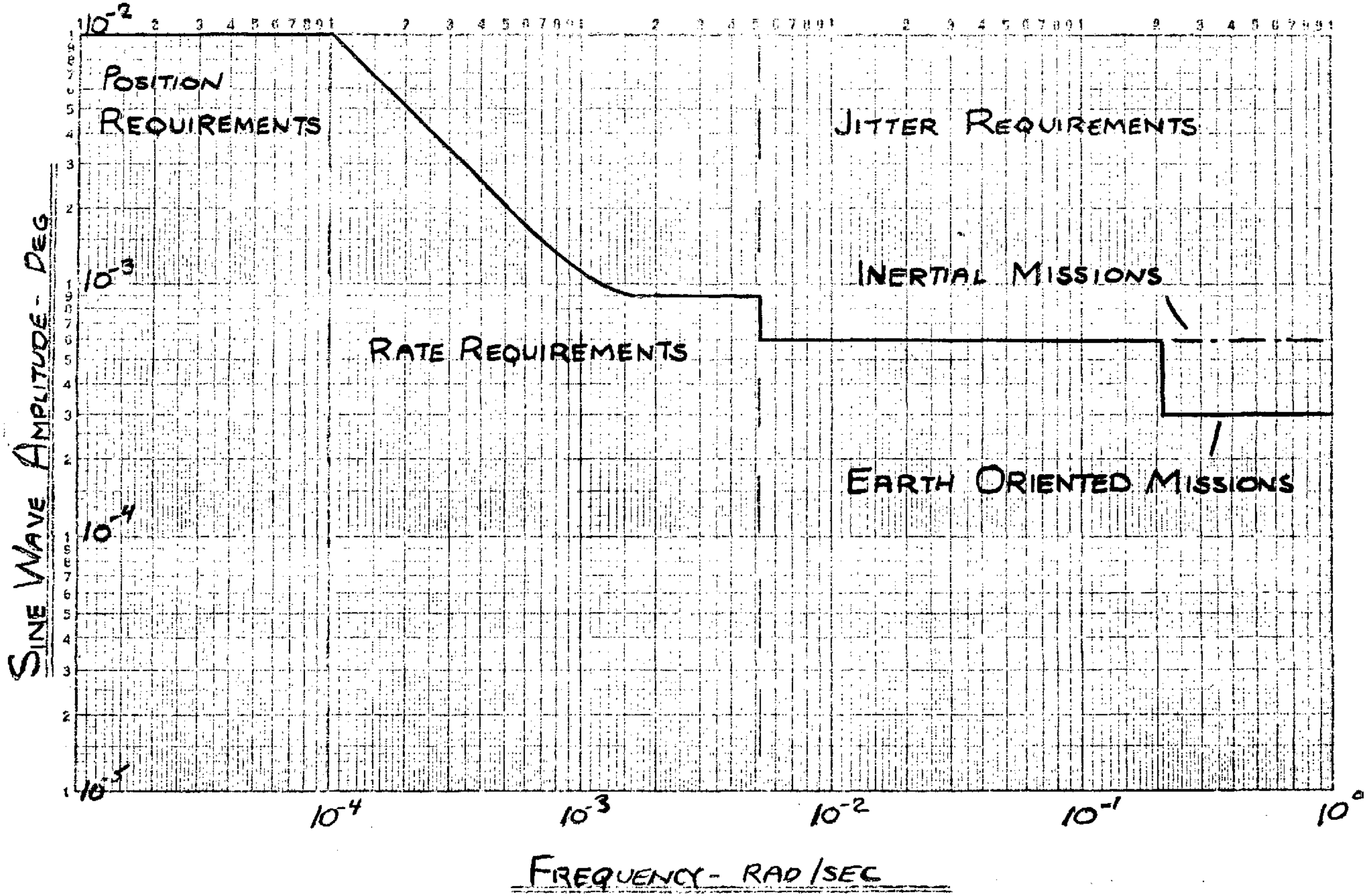


Figure 3-4. Spacecraft Attitude Requirements

3.2.1.1.6 Orbit Adjust/Maintenance

In the orbit adjust/maintenance mode, the subsystem shall control against thrust vector disturbance torques through use of reaction wheels, with reaction control jets enabled to unload wheel momentum or to backup wheel torque capability in a position dead band mode. For these maneuvers, control accuracy is 2.0 degrees orthogonal to and 2.0 degrees about the vehicle roll axis. Three axis reference and control accuracy shall be recovered within 60 minutes after completion of the orbit adjust/maintenance sequence. The orbit adjust torques will provide an acceleration equal to or less than 6.3×10^{-4} rad/sec^2.

3.2.1.2 Secondary Subsystem Performance Characteristics

3.2.1.2.1 Backup Momentum Management (low orbit missions only)

The subsystem shall be capable of unloading the momentum wheels using jet thrusters. The attitude error during the maneuver shall not exceed 0.1 degrees.

3.2.1.2.2 Backup Control to Sun Line

In the absense of software control capability, the subsystem shall orient any of the three digital sun sensor null axes to the sun line to within 0 $\pm$5 degrees. In the absence of the sun (earth occultation), the subsystem shall control vehicle attitude in a backup inertial hold mode to $\pm$.5 degrees over 30 minutes with capability for automatic transfer of control from the sun to inertial hold as a function of sun availability. Control about the sun line shall be rate limited with maximum drift rates of $\pm$0.5 degrees per hour.

3.2.1.2.3 Survivability

The subsystem shall be capable of providing attitude control about the sun line such that an operational solar array drive mechanism can orient the array within 7 degrees of the sun normal for 30 days after receipt of a sun hold mode command (primary or backup).

3.2.1.3 Attitude Control Loop Requirements

All sequence switching from one mode to another will be accomplished by ground commands. The control loops and modes are defined in the following subparagraphs.

3.2.1.3.1 Acquisition Modes

These modes shall result in the acquisition of a three axis reference based on software processing of available target stars.

3.2.1.3.1.1 Control to Sun Line

This mode shall result in pointing of the negative yaw axis to the sun from any initial attitude and from initial rates of up to ±3.0 deg/sec about any of all control axes, to an accuracy of 0 ± 1.0 degrees. Control axis rates shall be reduced to within 0 ±0.5 degrees/sec on all axes. The control loops shall operate in a continuous momentum management mode through software processing of rate data from the IRU and position data from the digital sun sensors. Software logic in sun acquisition shall be as follows:

a. Rate Reduction: Software shall develop low thrust jet command signals about each control axis as a function of spacecraft for rates above .2 deg/sec and null attitude about the sun line in excess of ±20 degrees.

b. Sun line acquisitions: Software shall develop jet command signals about each control axis according to the phase plane plot of Figure 3-5, in the rate limited region. Software shall develop wheel command signals about each control axis according to the null region shown in Figure 3-5, with jet control signals developed as a function of wheel speed. Equivalent position deadband shall be .5 ±.05 degrees for wheel control and ±3 degrees for jet control. Rate to position gain shall be 15. Acquisition field of view shall be such that the sun null sequence will take place with no more than one sun search rate command.

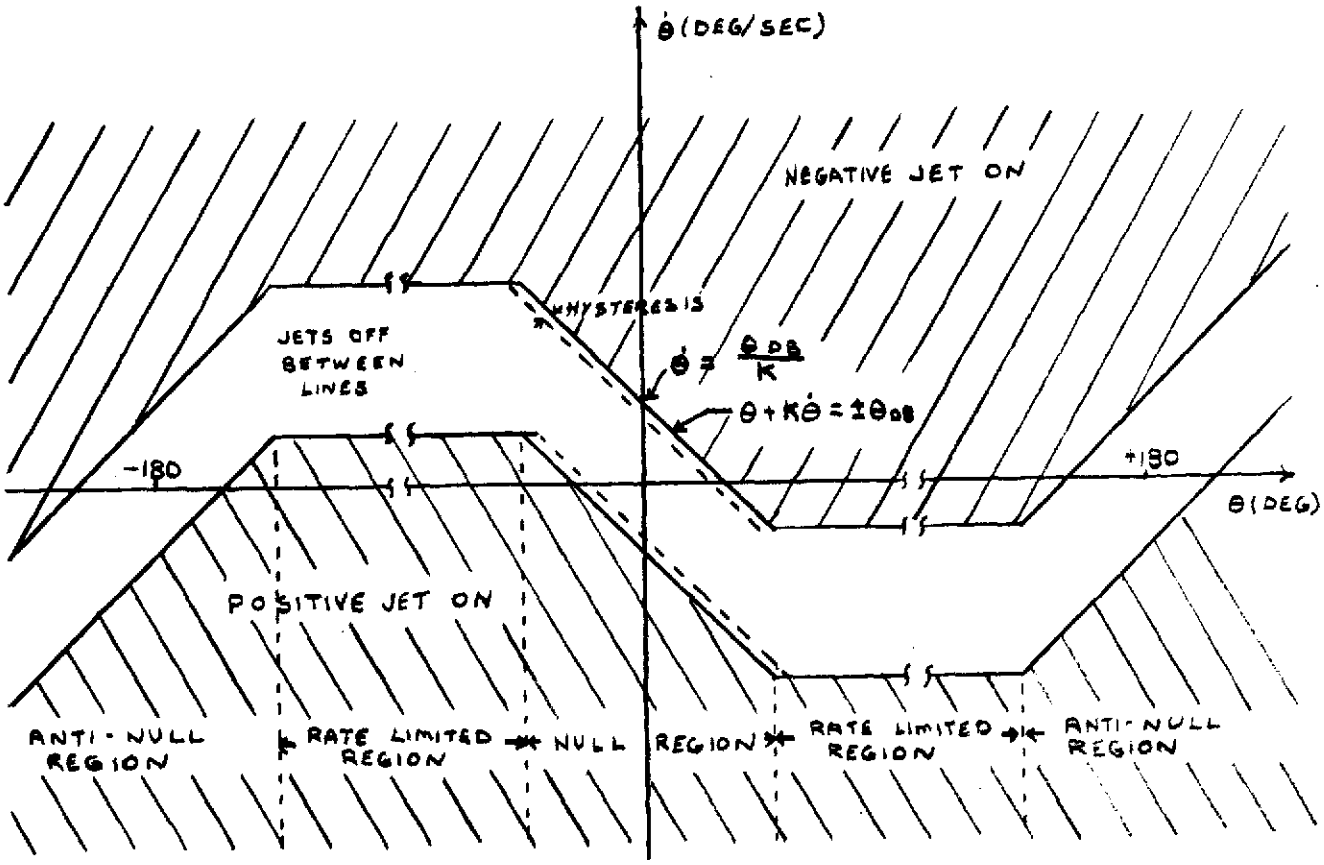


Figure 3-5. Phase Plane Plot

3.2.1.3.1.2 Star Search

This mode shall result in the acqusition of predetermined target stars by maintaining a constant spacecraft rate about the sun null axis defined in Paragraph 3.2.1.1.3. Software shall develop control commands necessary to establish a constant positive vehicle yaw rate about the sun line of 0.2 $\pm$.05 deg/sec. In addition, the star sensor shall be interrogated by software to accumulate sufficient target star data for initial reference acquisitions. Control in pitch and roll shall be through sun sensor data processing.

3.2.1.3.2 Operational Modes

These modes shall result in pointing the spacecraft positive yaw axis to an accuracy specified by the mission profile. Equipment used in these modes shall be the star sensors, on-board processing software, the IRU, reaction wheels and magnetic torquers.

3.2.1.3.2.1 Earth Pointing

This mode shall result in pointing the spacecraft yaw axis to the earth's centroid with an accuracy of 36 arc-seconds (1 sigma) per axis continuously. The influence of spacecraft ephemeris and star ephemeris in maintaining the software reference necessary to achieve this accuracy will be uplinked as required and will produce an attitude error no greater than 3.3 arc-seconds (1 sigma). Momentum management for low orbit altitudes (but greater than 300 n-mi) shall be accomplished by on-board processing of vehicle ephemeris in conjunction with a stored model of the earth's magnetic field and vehicle momentum state. Momentum management at synchronous orbit altitudes shall be accomplished by shaping wheel torque commands in conjunction with jet unloading pulses.

3.2.1.3.2.2 Inertial Hold

This mode is identical to the earth ponting mode except the reference frame is inertial and spacecraft ephemeris is irrelevant.

3.2.1.3.2.3 Orbit Adjust/Maintenance

In this mode, the subsystem will be directed to point the vehicle positive roll axis along a pre-determined vector in inertial space for the purpose of firing high thrust engines. Subsystem software shall process star sensor and IRU data to control vehicle slew rate in an eigenvector orientation maneuver at $.2 \pm .05$ deg/sec to the commanded attitude. During high thrust engine firing, the subsystem shall develop commands to drive low or medium thrust Jets (set by ground software) in an attitude deadband of 1 degree, with rate damping equivalent to 10:1.

3.2.1.3.1 Backup Modes

Table 3-1 describes control sensor/actuator backup mode capability that shall be provided through modification of on-board software in conjunction with logic commands received from uplink data.

Table 3-1. Backup Mode Capability

Failure	Alternate Sensors	Components Actuators	Performance Capability
Reaction Wheels	None	Low Thrust Jets	Position: TBD Rate: TBD
Magnetic Torquers	None	Low Thrust Jets	No degradation except when wheels are unloaded
Primary Sun Sensor	Use one of 2 remaining sensors	None	Must be capable of acquiring a celestial reference.
IRU	Use backup IRU	None	No degradation.
Star Sensor	Sun Sensor	None	Holds to sun line only.
AOP	Sun Sensor	Wheels & Jets	Holds to sun line only.

3.2.1.3.1.1 Failure Detector Mode

Upon receipt of a ground logic level, the subsystem shall interrupt all software actuator command data and respond to actuator command data from the backup controller. This mode will continue until ground commands reset the primary mode logic.

3.2.1.3.1.2 Backup Wheel Unloading

Upon receipt of a ground logic level, subsystem software shall develop wheel torque command signals for the purpose of impulse wheel unloading in the event of magnetic torquer failures. Vehicle rates shall be less than $3 * 10^{-4}$ deg/sec within 10 seconds after initiation of a .1 ft #-sec unload pulse about any axis.

3.2.1.3.1.3 Backup Control to Sun Line

The subsystem shall be capable of orienting either the spacecraft negative yaw, positive roll or negative roll axis to the sun line for a period not less than 30 days measured from the time the mode is commanded. Control sensors and actuators can be made up of any compatible combination of available hardware. Control to the sun line shall be within ± 3 degrees and rates about the sun line shall be $0 \pm .05$ degrees/sec.

3.2.1.4 Component Performance Characteristics

3.2.1.4.1 Inertial Reference Unit

The Inertial Reference Unit (IRU) shall provide short term orthogonal axis reference update information to the inertial reference frame maintained in on-board processor software. The information is supplied in serial binary form when interrogated by the remote multiplexer. In a backup mode, the IRU shall supply analog rate information to the backup controller.

3.2.1.4.1.1 Rate Capability

In high gain mode, the IRU shall be capable of linearly accumulating rate data over the range of ± 3 deg/sec about each of its sensitive axes. In low gain mode maximum rate capability shall be 1.2 deg/sec per axis. Least significant bit granularity shall not exceed 0.9 arc-sec and 0.06 sec respectively in high and low gain mode.

3.2.1.4.1.2 Random Drift (3 sigma) $1.4 * 10^{-4} < f < .1$

In high gain mode, uncertainty in rate drift per axis in the above specified frequency band shall not exceed 0.045 deg/hr. In low gain mode this drift uncertainty shall not exceed 0.003 deg/hr. Uncalibrated drift per-axis in either mode shall not exceed .5 deg/hr. Calibration can be assumed to be periodic at 600 second intervals throughout gyro useful life.

3.2.1.4.1.3 Scale Factor Stability

Uncertainty in scale factor shall be less than .03% after on-orbit calibration due to ageing, variations in temperature calibration curves, and variation in calibrated response to magnetic fields.

3.2.1.4.1.4 Gyro Dynamics

Each sensitive axis captive loop electronics shall be set such that output response to a step input in rate will correspond to that of a critically damped (ζ =.7$\pm$.05) second order quadratic of 3 HZ $\pm$ .5%.

3.2.1.4.1.5 Rate Noise $F > 0.1HZ$

When placed in the above caging loop, RMS noise at the output of each axis about 0.1 HZ shall not exceed $1.5 * 10^{-3}$ deg/sec in high gain mode and 10^{-4} deg/sec in low gain mode.

3.2.1.4.2 Star Sensor

This component shall supply spacecraft control information for the purpose of re-setting the software attitude reference frame based on a pre-determined star availability pattern. With a star of proper magnitude in the sensor field of view, two digital signals representing the angular displacement about axes orthogonal to the sensor boresight axis will be available for interrogation.

3.2.1.4.2.1 Field of View

This component shall have a useable square field of view not less than ±4 degrees, centered with respect to the optical boresight axis. The component shall be capable of initiating and re-initiating (based on external command) a star search in this field of view for acquisition of target stars. The sensor field of view while tracking a target star shall be no greater than .15 deg half cone angle.

3.2.1.4.2.2 Star Sensivity

This component shall be capable of detecting stars of magnitude brighter than +6 based on threshold level commands received from on-board software. Calibrated star magnitude sensitivity shall be such that stars can be identified within ± 1 magnitude.

3.2.1.4.2.3 Tracking Accuracy

While tracking a star, processed data readout with respect to the sensor boresight axis shall be accurate to within 10 arc sec per axis (1 sigma). Calibration curves required to produce this accuracy over the sensor field of view shall take no more than TBD computer memory, in conjunction with an interpolation algorithum. This accuracy shall be provided for all target stars brighter than +6 visual magnitude.

3.2.1.4.2.4 Noise Equivalent Angle

The component shall provide a noise equivalent angle for +6 magnitude stars no greater than 3 arc-seconds (1 sigma) per axis for a 1 rad/sec tracking mode bandwidth.

3.2.1.4.2.5 Stray Light

This component shall meet the performance specified in paragraphs 3.2.1.1 through 3.2.1.3 when its sensor boresight axis comes as close as 30 degrees to the limb of any bright object (sun , earth, or moon). Protective devices (sun shutters) , if necessary, shall be activated as part of the component internal control logic to protect against component damage.

3.2.1.4.3 Sun Sensor

This component shall provide in the form of an analog signal, information relative to the direction of the sun from the spacecraft. This component shall consist of three detector heads each using two orthogonal sensors, and associated electronics. In addition, this component shall present analog sun angle data related to each detector lead for the purpose of backup attitude control to the sun line.

3.2.1.4.3.1 Field of View

Each sensor head shall be capable of responding to sun angles with respect to its insensitive axis over a square field of view not less than ±64 degrees.

3.2.1.4.3.2 Resolution

The digital or analog data from each head will appear as a staircase. The resolution of each step (LSB) shall be no greater than 0.5 deg per axis.

3.2.1.4.3.3 Accuracy

Calibration parameters shall be provided with the sensor such that two axis angle data with respect to the sensitive axes can be recovered in closed form through use of software external to the component. Processed data accuracy shall be within 0.25 deg (excluding the influence of resolution) for any sun angle in the sensor field of view.

3.2.1.4.4 Momentum Wheel

This component shall provide the reaction torques necessary for control of spacecraft attitude during pointing modes. The flywheel shall be an integral part of a motor used to develop the reaction torques. Three identical wheels will be used, such that control torques about each vehicle axis can be commanded independently. Control torques will be developed through application of 400 HZ 2 phase power supplied from the Drive Electronics. Torque level will be set as a function of percent duty cycle in the applied fixed and control phase voltage square waves and torque sense will be set as a function of lead vs. lag in the control phase voltage with respect to the fixed phase voltage.

3.2.1.4.5 Momentum Capability

Each wheel shall be capable of storing 7.0 ft-#-sec of momentum at wheel speeds not to exceed 3000 RPM.

3.2.1.4.5.1 <u>Control Torque Capability</u>

Each wheel shall be capable of acceleration torques up to 20 in-oz at 100% command duty cycle (fixed and control phase). Control torque at constant command duty cycles above 10% shall be constant $\pm$15% for wheel speeds from zero to two thirds of no load speed. Control torque as a function of command duty cycle above 10% shall be linear $\pm$ 10%.

3.2.1.4.5.2 <u>Tachometer</u>

This component shall supply pulse data to the Drive Electronics. Pulse frequency shall be one or more pulses per wheel revolution and pulse polarity shall reverse as a function of wheel rotation sense.

3.2.1.4.6 Magnetic Torquer Assembly

This component shall supply magnetic moments about three orthogonal axes with magnitude linear as a function of analog voltage supplied by the Drive Electronics.

3.2.1.4.6.1 Capacity

Magnetic moment capacity about each control axis shall be no less than 30,000 pole-cm when 28 volts are supplied across the input lines of the torquer.

3.2.1.4.6.2 Hysteresis

Pole strength shall be linear as a function of applied voltage $\pm$ 300 pole-cm for increasing and decreasing command inputs. Pole strength per axis for zero command input shall be no greater than 300 pole-cm.

3.2.1.4.7 Drive Electronics

This component shall be capable of driving 12 jet control valves as part of the PRCS interface. In addition this component shall apply linear voltage to three magnetic torquers and pulse width modulation control of AC voltages to three reaction wheels. The actuator signals above will be the result of response in this component to command and logic signals from the on-board processor (normal mode) or the backup controllers (computer failure mode). Tachometer pulse data from each of the three reaction wheels shall be processed in the Drive Electronics to develop analog signals with magnitude representing stored momentum.

3.2.1.4.7.1 Wheel Driver Electronics

This component shall process the computer clock (1.6mHZ) such that 400HZ, 2 phase control square waves can be used to gate regulated power to each of three reaction wheels. The gating interval shall be separately controlled through processing of digital command words (D/A and sample + hold) representing percent gating duty cycle, in addition to a sign bit that controls the phase relationship between the 2 phase 400 HZ power delivered. For backup control of wheels, this component shall accept a level to inhibit software data received, and output pulse modulated square waves as a function of analog signals, representing percent duty cycle, from the backup controller.

3.2.1.4.7.2 Jet Driver Electronics

Twelve separate electronic switches shall independently control current to twelve jet solenoid valves as a function of logic levels set and reset by computer software through the remote decoder. The state of each driver shall be held by the set/reset level commanded at discrete intervals from the on-board computer. In addition, when commanded by an inhibit logic level the state of eight of the twelve drivers shall be controlled by logic levels from the backup controller.

3.2.1.4.7.3 Magnetic Torquer Drive Electronics

The component shall process three separate digital commands (D/A and sample + hold) representing percent of maximum di-pole moment capacity to each of three magnetic torques. The control signals shall unbalance a linear power bridge such that bi-directional voltage levels equivalent to the specified pole strength will be available across each torquer coil.

3.2.1.4.8 Backup Controller

This component will be commanded as the result of an on-board processor malfunction, for the purpose of supplying wheel and jet torque commands to the Drive Electronics. The output signals shall be developed from analog rate data supplied by the IRU and analog position data supplied by the digital sun sensor electronics.

3.2.1.4.8.1 <u>Sun Sensor Data</u>

Two axis analog data from each of three sun sensor heads will be available for processing as a function of the logical combination of sun presence signals from each of the sensors. Threshold levels shall be set in the logic such that the selected head for initial sun acquisition is held throughout the acquisition sequence. Each analog sun sensor channel shall be processed such that actuator control torques will be developed from the signals that reduce errors to a sun null with respect to that sensor. Four of the six error signals shall be grounded as a function of the sun presence control logic.

3.2.1.4.8.2 Inertial Reference Unit Data

Analog rate data from all channels of the IRU will be available for rate damping when the sun is present, and for rate + position control when no sun is present. Selection of rate channels and rate + position channels shall be per sun presence logic in conjunction with ground command (inhibiting redundant IRU channels).

3.2.1.4.8.3 Signal Processing

The selected position and rate signals shall be combined to supply analog bi-directional signals representing percent wheel torque for use in the Drive Electronics. When position plus rate signals exceed a level equivalent to ± 3 degrees, a logic level shall be set for controlling the respective valve driver in the Drive Electronics. Jet valve logic should be such that jet on-times shall be 50 ms $\pm 10\%$ and jet off times shall be 10 seconds. For rate plus position signals that exceed a level equivalent to 20 degrees, valve on-time shall scale analog signals representing wheel momentum such that jet valve logic associated with the 3 degree threshold shall fire unloading jet pulses per position deadband pulse above.

3.2.2 PHYSICAL CHARACTERISTICS

All subsystem hardware shall be mounted in the ACS module except for the plus and minus roll digital sun sensor heads. Physical characteristics of the subsystem are specified in the following paragraphs.

3.2.2.1 Size

All components mounted in the ACS module shall be located consistent with paragraphs 3.1.2.2 and paragraph 3.2.2.4 within the module envelope 40" long x 48" wide x 16" high. In addition to subsystem hardware, the module shall accommodate the remote decoder/multiplexer whose dimensions are 4" x 5" x 2", and subsystem interface harness. Component size requirements will not be controlled by this specification, but appear in Table 3-2 for reference.

3.2.2.2 Weight

The total weight of this subsystem (including external digital sun sensor heads) shall not exceed 137 pounds. Component weight requirements will not be controlled by this specification, but appear in Table 3-2 for reference.

3.2.2.3 Power

The total peak power demand of this subsystem after component turn-on transients have settled shall not exceed 400 watts, and orbit average power demand of this subsystem shall not exceed 50 watts. Component power requirements will not be controlled by this specification, but appear in Table 3-2 for reference.

Table 3-2. Component Physical Characteristics

Item	Quantity	Size	Weight (#)	Power (W)	Sensitive Axis Alignment Accuracy
Momentum Wheel	3	13" dia x9	12.2	3	$\pm 1^{\circ}$
Star Sensor	1	5¼"x6"x12"	11	5	TBD
Magnetic Torquers	3	1.7" diax15"	3.3	1	$\pm 1^{\circ}$
Inertial Reference Unit	1	12"x4.5"x7"	12.0	22	$\pm .1^{\circ}$
Sun Sensor Electronics	1	3.5"x4.5"x1.2"	3.7	.5	-
Sun Sensor Heads	3	3.2"x3.2"x.8"	.1	-	$\pm .1^{\circ}$
Drive Electronics	1	6"x6"x8"	10	5	-
Backup Controller	'	6"x6"x4"	5	5(not in normal mode)	-
Harness	1	-	7	-	-
Module Structure	1	40"x48"x16"	40	-	-

3.2.2.4 Component Alignment

Component mounting capability to the subsystem module shall allow for adjustment as specified in Table 3-2. Reference cubes or alignment aids shall be supplied as required by each component to obtain the measurement accuracy with respect to the module reference surface specified in Table 3-2.

3.2.3 RELIABILITY

The probability of meeting the performance requirements of this specification for the various operational modes shall be no less than that indicated below, assuming that each mode starts at spacecraft separation from booster and continues for the number of hours indicated:

Mode	Hours	Reliability
Initial Acquisition	TBD	TBD
3 Re-acquisitions	TBD	TBD
Operational Control (normal mode)	17,520	TBD
Orbit Maintenance/Orbit Adjust	1,900	TBD
Backup Control	720	TBD

3.2.4 MAINTAINABILITY

3.2.4.1 Maintenance Requirements

This subsystem shall be designed for ease of maintainability to minimize equipment downtime during assembly, test, and checkout.

3.2.4.2 Maintenance and Repair Cycle

With the exceptions noted below, the design of this subsystem shall be such that no scheduled maintenance will be required.

a. During ground phases, critical optical surfaces will be periodically inspected and cleaned if necessary, prior to subsystem use.

b. During ground phases, the momentum wheels will be operated and performance verified at a minimum of 6 month intervals and subsequently stored with the spin axis in a horizontal position.

c. During ground phases, the inertial reference until shall be operated for a minimum of two hours, once every three months, with subsequent storage of the components in a randomly-chosen position.

3.2.4.3 Service and Access

Access to individual components in the ACS module shall be provided for inspection, servicing, and replacement without requiring major disassembly of the subsystem. Removable dust covers or other protective devices shall be provided over critical optical elements and shall be easily removed prior to subsystem use. Quick-disconnect electrical connectors shall be utilized to facilitate component removal. Provisions shall be made to verify operation of redundant paths without component disassembly, except for paths consisting of non-switched parallel conductors.

3.2.4.4 Useful Life

The useful life of this subsystem shall be a minimum of 2 years starting with the acceptance of the spacecraft by the procuring agency. The useful life of individual components within this subsystem shall include additional time accrued during the transportation, handling, storage, and testing phases prior to acceptance by the procuring agency.

3.2.5 ENVIRONMENT CONDITIONS

This subsystem shall be such that it will meet all performance requirements stated in Section 3 of this specification after the subsystem is subjected to the transportation environments described in 3.2.6.

3.2.6 TRANSPORTABILITY

Later.

3.3 DESIGN AND CONSTRUCTION

3.3.1 MATERIALS, PROCESSES, AND PARTS

3.3.1.1 Selection of Materials, Processes and Parts

Materials and processes shall be selected in accordance with the requirements of TBD.

3.3.1.2 Selection of Electronic Parts

Electronic parts shall be selected in accordance with the requirements of PPL-12, latest issue.

3.3.1.3 Screening of Parts

Parts shall be screened in accordance with the requirements of TBD.

3.3.1.4 Parts Specifications

Parts specifications shall be prepared in accordance with TBD.

3.3.1.5 Part Application Restrictions

The application restrictions defined in TBD.

3.3.1.6 Parts Derating

Parts shall be derated in accordance TBD.

3.3.1.7 Traceability of Parts

The manufacturer's part number, lot number, and date code of all electronic parts assembled into prototype or flight equipment shall be recorded. Each part assembled to printed circuit boards shall have its identification markings visible after assembly.

3.3.1.8 Corrosion Prevention

The use of dissimilar metals as specified in MIL-STD-454, Requirement 16, shall be avoided wherever possible. Materials, techniques, and processes shall be selected and employed with regard to heat treatment procedure, corrosion protection, finish and assembly and installation such that sustained or residual surface tensile stress, stress concentrations, and the hazards of stress corrosion, cracking, and hydrogen embrittlement are minimized. Processes and materials for protection against corrosion of metal parts shall be selected from those specified in paragraph 3.3.1.1, with the exception that cadmium plating shall not be used. Selected finishes shall be compatible with the thermal requirements of this specification. All parts shall be corrosion resistant or have a suitable protective coating applied.

3.3.1.9 Moisture and Fungus Resistance

Materials which are not nutrients for fungus and which resist damage from moisture shall be used wherever possible. The requirements of MIL-STD-454, Requirement 4, shall apply. The use of materials which are nutrients for fungus are not prohibited in hermetically sealed assemblies and in other accepted and qualified uses; such as paper capacitors and treated transformers. If it is necessary to use fungus nutrient materials in other than such qualified applications, these materials shall be treated with a process which will render the resulting exposed surface fungus resistant.

Protective coatings shall not be acceptable as moisture and fungus preventatives for parts which may lose their coatings during normal course of assembly, inspection, maintenance and testing.

3.3.2 ELECTROMAGNETIC COMPATIBILITY

Electromagnetic compatibility shall be in accordance with the detailed requirements of GE Specification SVS-7997, Electromagnetic Compatibility Requirements for Components and Subsystems.

3.3.3 NAMEPLATES AND PRODUCT MARKING

The component shall be marked for identification in accordance with the manufacturer's standards. The identification shall include, but not be limited to, the following:

1. Nomenclature
2. Customer Part Number
3. Serial Number (Engineering models will use a different designation than prime hardware)
4. Contract Number
5. Manufacturer's Name or Trademark
6. Date of Manufacture (month, day, year)
7. Property: "NASA"

Hardware or equipment which is not suitable for use in flight, and which would be accidentally substituted for Flight or Flight Spares Hardware shall be red striped with material compatible red paint to prevent such substitution. In the event the hardware is too small to be easily striped, or it test results would be affected by striping, a conspicuous red tag marked "NOT FOR FLIGHT USE" shall be attached.

Wire and Cables. Wires and cables for hardware shall not be identified by hot stamping directly onto primary or secondary (shield) insulation.

3.3.4 WORKMANSHIP

The component, including all parts and subassemblies, shall be constructed, finished and assembled in accordance with the highest standards for high reliability aerospace equipment. Workmanship criteria shall comply with MIL-STD-454, Requirements 9 and 24. Particular attention shall be paid to neatness and thoroughness of soldering, wiring, marking of parts and assemblies, plating, painting, machine screw assemblage and freedom from burrs and sharp edges.

3.3.5 CLEANLINESS

Hardware shall be designed, manufactured, assembled, and handled in a manner to insure the highest practical level of cleanliness. Suitable precautions shall be taken to insure freedom from debris within the hardware, and inaccessible areas where debris and foreign material can become lodged, trapped, or hidden shall be avoided. Hardware shall be designed so that malfunctions or inadvertent operating cannot be caused by exposure to conducting or nonconducting debris or foreign material floating in a gravity-free state. Electrical circuitry shall be designed and fabricated to prevent unwanted current paths being produced by such debris. Ultrasonic vibration shall not be used as a method for cleaning component electronic assemblies.

3.3.6 INTERCHANGEABILITY

Each subassembly of the component shall be interchangeable with regard to form, fit and function with other subassemblies of the same part number. Likewise, the component itself shall be directly interchangeable with other serial number components. The requirements of MIL-STD-454, Requirement 7 shall apply.

3.3.7 SAFETY PRECAUTIONS

Warnings and precautions relative to personnel and equipment safety shall be specified in component handling, assembly and test instructions.

SECTION 8.0

SPECIFICATION SVS XXXX

16 SEPT. 1974

SPECIFICATION

FOR THE

EARTH OBSERVATORY SATELLITE (EOS)

HYDRAZINE REACTION CONTROL SUBSYSTEM

TABLE OF CONTENTS

TABLE OF CONTENTS (Cont'd)

1.0 SCOPE

This specification establishes the performance, design, and test requirements for a Reaction Control Subsystem, hereinafter referred to as the RCS. The RCS is a monopropellant (hydrazine) fueled, varying thrust (blowdown) rocket engine system which is used to perform various attitude control functions for an earth orbiting spacecraft.

2.0 APPLICABLE DOCUMENTS

The following documents (of the exact issue shown) form a part of this specification to the extent specified herein. In the event of conflict between documents referenced here and the detail requirements of Sections 3, 4, and 5, the detail requirements of Sections 3, 4, and 5 shall supersede. In the event of conflict between documents referenced here and lower tier references in documents referenced here, the former shall supersede.

Military

MIL-P-26536C	Propellant Hydrazine
MIL-P-27401B	Propellant Pressurizing Agent, Nitrogen
MIL-A-18455B	Argon, Technical
MIL-P-27407 Suppl. I	Propellant, Helium, Pressurizing
MIL-STD-454C	Standard General Requirements for Electronic Equipment

Federal

TT-I-735	Isopropyl Alcohol

NASA

NHB 5300.4 (3A) May 1968	Soldering of Electrical Connectors

Air Force

AFWTRM -127-1	Air Force Western Test Range Safety Manual

General Electric

TBD	Electromagnetic Compatibility Requirements for Components and Subsystems
TBD	Environmental Design and Test Requirements for Components and Subsystems
TBD	Approved Materials and Processes List
TBD	Electrical System Interface Requirements
TBD	Harness Design Requirements
TBD	Approved Parts List

DRAWINGS

General Electric

TBD	Propulsion Module Envelope Drawing
TBD	Propulsion Module Structural Assembly

3.0 REQUIREMENTS

3.1 ITEM DEFINITION

The RCA is a monopropellant hydrazine type propulsion system of single module construction consisting of a propellant storage and expulsion section, a propellant distribution section, and a rocket engine section. The RCS provides the mass expulsion used for performing the following attitude control functions:

(1) Initial stabilization of the spacecraft

(2) Reaction torque to counteract the torque produced during momentum wheel unloading

(3) Restabilization of the spacecraft to the celestial references

(4) Limit cycle attitude control of the spacecraft

3.1.1 ITEM DIAGRAMS

The RCS block diagram defining the flow schematic and component location is shown in Figure 1. The RCS shall be designed as a single module of all welded of brazed construction (upstream of all engine valve seats) and shall be capable of spacecraft installation as a completely assembled subsystem (without propellant and pressurant) on the RCS propulsion module structure.

3.1.2 INTERFACE DEFINITION

3.1.2.1 Power

All electrically operated components of the RCS shall operate from a power source having characteristics as defined in paragraph 3.0 of General Electric Document (TBD)

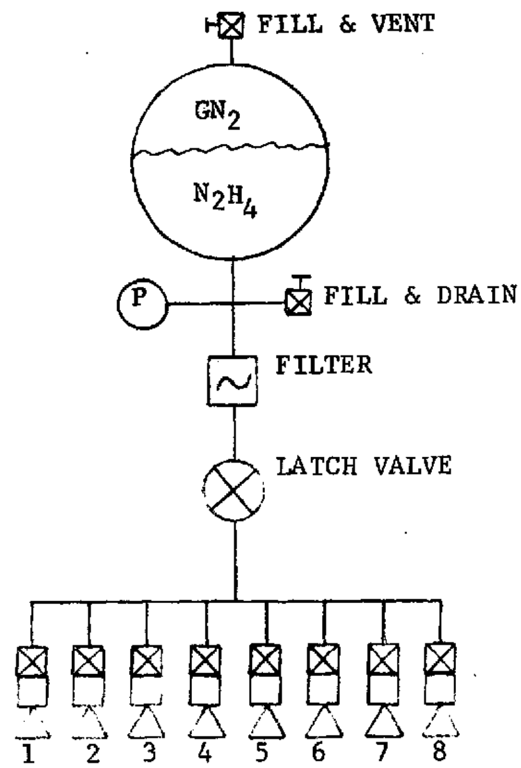
3.1.2.2 Telemetry

The RCS shall provide for telemetry measurement points at locations identified in Table 1. Telemetry measurements shall have interface characteristics as defined in Paragraph 5.0 of General Electric Document (TBD). Pressure sensor calibration data provided by the subcontractor shall be a minimum of five (5) data points plotted in newtons/sq.cm absolute versus volts. Temperature sensor calibration data provided by the subcontractor shall be a minimum of five (5) data points plotted in degrees centigrade versus volts. The pressure sensor calibration data shall have an accuracy of $\pm 1\%$ of the full scale measurement range. The temperature sensor calibration data shall have an accuracy of $\pm 5.5^{\circ}C$ through a measurement range of $0^{\circ}C$ to $260^{\circ}C$ and shall have a measurement capability of from $-29^{\circ}C$ to $955^{\circ}C$.

3.1.2.3 Thermal Control

The RCS, exclusive of the rocket engines, shall be thermally controlled during orbit to the limits specified in paragraph 3.1 of GE specification (TBD). The RCS shall incorporate the required rocket engine electrical heaters in order to meet the performance requirements of paragraph 3.2.1 under the environmental conditions of deep space. These conditions shall include the seasonal variation

Figure 1. Reaction Control Subsystem Block Diagram



REA - 0.127 Kgf

FUNCTION	REA
+ Roll	1 and 5 or 3 and 7
- Roll	2 and 6 or 4 and 8
+ Pitch	3 and 8
- Pitch	4 and 7
+ Yaw	2 and 5
- Yaw	1 and 6

of the solar flux as well as nominal tolerances as follows:

Solor Flux S_o = 1353 ± 13.5 Watts/m^2

Seasonal Variation = +46.4, -44.1 Watts/m^2

Rocket engine electrical heaters shall be controlled by Ground Command.

3.1.2.4 Structure

The RCS shall be assembled on structure shown on GE drawing (TBD). The RCS module space envelope and rocket engine orientation shall be as defined on GE drawing (TBD). The RCS module shall withstand the static and dynamic environments of paragraph 3.1 of GE specification (TBD).

Table 1. Reaction Control Subsystem

Telemetry Measurements Points

Propellant Feed Pressure
Propellant Tank Temperature
REA Chamber Temperature (1)
REA Chamber Temperature (2)
REA Chamber Temperature (3)
REA Chamber Temperature (4)
REA Chamber Temperature (5)
REA Chamber Temperature (6)
REA Chamber Temperature (7)
REA Chamber Temperature (8)
Latch Valve Position

3.1.2.5 Aerospace Ground Equipment

The RCS shall provide for interfaces with the following items of AGE:

(1) Propellant and Pressurant Servicing Cart

(2) Rocket Engine Nozzle Alignment Targets

(3) Subsystem Electrical Test Set

(4) Subsystem Shipping Container

3.1.3 MAJOR COMPONENT LIST

The RCS is composed of the following components:

(1) Low Thrust Engine (8)

(2) Propellant Tank Assembly (1)

(3) Fill and Drain Valves (2)

(4) Latching Valves (1)

(5) Propellant Filter (1)

(6) Pressure Transducer (1)

(7) Electrical Harness (1)

(8) Support Structure (1)

3.2 CHARACTERISTICS

3.2.1. PERFORMANCE

3.2.1.1 Low Thrust Engines

The RCS Low Thrust Engines (LTE) shall have the following performance characteristics.

3.2.1.1.1 Thrust Level. Each LTE shall provide an initial thrust of 0.127 $\pm$ 10% Kgf. This thrust level shall be achieved under initial tank pressure conditions and at vacuum using 20°C propellant and pressurant. The nominal thrust level upon completion of the mission shall exceed 0.03 Kgf.

3.2.1.1.2 Total Impulse. The RCS shall provide a minimum total impulse of 1215.0 Kgf-sec for accomplishing spacecraft functions. Any single LTE shall have the capability of providing 750 Kgf-sec of total impulse.

3.2.1.1.3 Duty Cycle. In any single operation at initial tank pressure conditions, each LTE shall be capable of a minimum impulse burn of .002 Kgf-sec ($\approx$ 7 ms pulse) and a maximum impulse burn of 10 Kgf-sec. Each LTE shall be capable of 100,000 on-off cycles. Of these, 20,000 cycles shall be initial starts. The LTE's shall not be duty cycle limited.

3.2.1.1.4 Predictability. The impulse predictability for each LTE shall be as shown in Figure 2.

3.2.1.1.5 Specific Impulse. The steady state minus three sigma specific impulse for a LTE steady state burn at initial thrust level conditions and with the propellant at 20°C shall exceed 222 Kgf-sec/Kgm.

3.2.1.2 Engine Alignment

The actual thrust vector of each engine shall subtend an angle of $\leq \pm$ .15 degrees, 3σ variation, with the nozzle geometric centerline under all the operating conditions specified in paragraph 3.2.1. The alignment fixture tolerance (i.e., the uncertainty in knowledge of the nozzle geometric centerline angle with respect to the alighment mandrel mirror reference), shall be $\leq \pm$ .1 degrees, 3σ variation. An adjustment range allowing $\pm$ 2 degrees rotation of the nozzle geometric centerline about two (2) mutally perpendicular directions with an adjustment resolution of $\pm$ .05 degrees shall be provided in the design of each engine mount. The nozzle geometric centerline null shall be coincident with the axes as defined in General Electric (TBD).

3.2.1.3 Stability

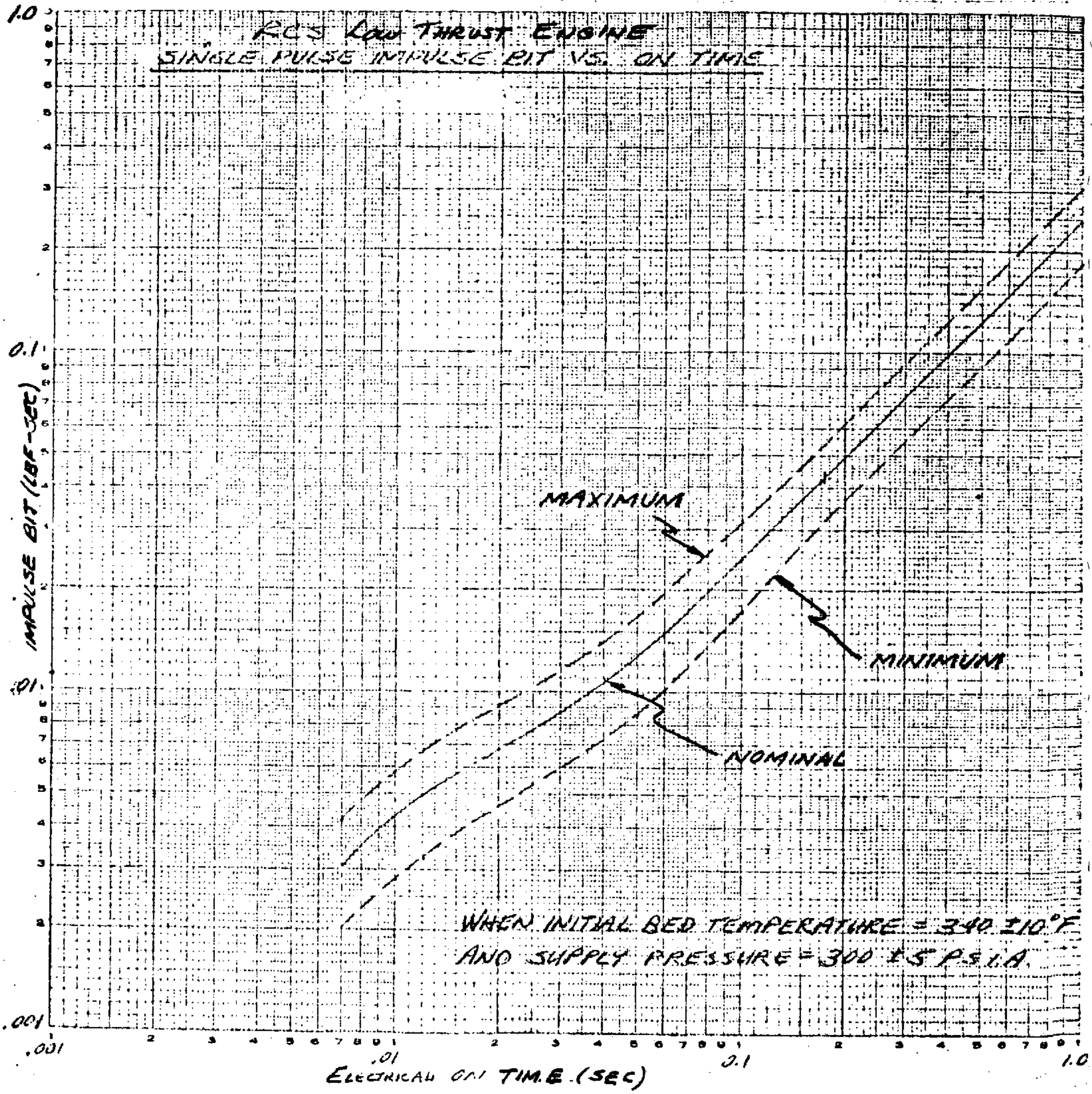
Engine chamber pressure oscillations occurring during an engine operation for the period starting with application of the power signal to the propellant control valve plus 0.1 seconds and ending with the removal of the power signals from the propellant control valve shall not exceed $\pm$ 25% of the nominal steady state chamber pressure.

3.2.2 PHYSICAL CHARACTERISTICS

3.2.2.1 Configuration

The RCS shall be designed in a modular arrangement of components. The RCS shall

Figure 2. Low Thrust Engine Impulse vs On Time



be assembled on a structure assembly (General Electric Drawing No. (TBD). All component interconnect tubing joints shall be welded or brazed.

3.2.2.2 Weight

The dry weight of the RCS shall not exceed 9.1 Kg. This weight does not include the RCS Structure Assembly. The RCS shall have the capability of being serviced with 13.6 Kg of hydrazine plus the required weight of nitrogen pressurant.

3.2.2.3 Size

The RCS size size shall be contained within the space envelope shown on General Electric Drawing No. (TBD).

3.2.2.4 Leakage Rate

The total external leakage rate of the RCS shall not exceed 30 SCC/hr of gaseous nitrogen when the RCS is subjected to system operating pressures.

3.2.2.5 Structural Pressures

The RCS shall be capable of withstanding a proof pressure of 1.5 times, and a burst pressure of not less than 4.0 times the maximum operating pressure, except the propellant tank, which shall have a burst to operating pressure ratio of 2 to 1 minimum at maximum operating pressure and maximum temperature.

3.2.2.6 Storage, Transporation, Handling, Assembly and Checkout

The RCS shall be capable of performance in accordance with 3.2 herein, after exposure to the applicable environments of paragraph 3.1 of GE Specification (TBD) or the following times:

Transportation	6 weeks
Storage	3 years
Handling and Assembly	3 months (system level)
Test and Checkout	6 months (system level)

3.2.2.7 Design Life

The design life of the RCS shall be a minimum of seven years starting with acceptance of the RCS by General Electric. The design life of individual components within the RCS shall include additional time accrued prior to incorporation into the RCS (i.e., transportation, handling, storage, and testing at the component level). The design life of the RCS shall consist of an accumulation of the preoperational phase of 3.2.2.6 followed by three years of operational life.

3.2.2.8 Cleanliness

To insure proper performance of the RCS, all components, and the subsystem itself, shall meet the cleanliness requirements of Table II. In addition, no metal particles shall be allowed which are over fifty (50) microns.

TABLE II

Propellant Particulate Cleanliness Requirements

Size Range (Microns)	Maximum Particles allowed per 100 Milliliter Sample
5-10	1200
11-25	200
26-50	50
51-100	5
over 100	0

3.2.2.9 Factors of Safety

The following factors of safety shall be applied to the maximum anticipated applied loads to obtain the respective limit loads and ultimate loads for design/analyses purposes.

Limit: 1.15 Ultimate: 1.25

3.2.2.9.1 Limit Loads. The component shall be designed for sufficient strength to withstand simultaneously the limit loads and other accompanying environmental phenomena given in GE document (TBD) for each design condition without experiencing yielding or excessive elastic deformation.

3.2.2.9.2 Ultimate Loads. The component shall be designed to withstand simultaneously the ultimate loads and other accompanying environmental phenomena without failure. Failure is defined as structural collapse, rupture, or other inability to sustain the ultimate loads.

3.2.2.10 <u>Wiring and Connectors</u>

All harness wiring and connectors shall be in accordance with General Electric Document (TBD).

3.2.2.11 <u>Dielectric Strength and Insulation Resistance</u>

There shall be no evidence of dielectric breakdown when the RCS harness insulation is subjected to 600 volts ac. Insulation resistance shall be a minimum of 50 megaohms when measured at 100 volts dc between all mutually insulated parts and ground.

3.2.3 RELIABILITY

The three-year probability of the RCS meeting the performance requirements of this specification shall be (TBD) Reliability apportionments for components of the RCS shall be determined by the propulsion subcontractor.

3.2.4 MAINTAINABILITY

The RCS shall be designed for ease of maintainability to minimize equipment downtime during assembly, test, and checkout.

3.2.4.1 Maintenance and Repair Cycles

The RCS shall be designed such that periodic maintenance will not be required.

3.2.4.2 Service and Access

Access shall be provided to the pressurant and propellant fill and drain valves and the electrical harness interface connectors. Provision shall be made to verify operation of redundant paths without RCS disassembly. Provisions shall be made to measure RCS external leakage and internal leakage of pneumatic components without RCS disassembly.

3.2.5 ENVIRONMENTAL CONDITIONS

The RCS shall be designed to withstand the environmental conditions specified in paragraph 3.1 of GE document (TBD) .

3.2.6 TRANSPORTABILITY

The design of the RCS shall be such that the subsystem will meet all performance requirements stated in Section 3 of this specification after the subsystem in its shipping container is subjected to the transportation environments described in GE document (TBD) . The RCS shall be designed for shipment by highway (common carrier) and/or aircraft. The RCS will not be serviced with propellant during transportation.

3.3 DESIGN AND CONSTRUCTION

3.3.1 MATERIALS, PROCESSES AND PARTS

3.3.1.1 Selection of Electronic Parts

Electronic parts shall be selected in accordance with GE Document (TBD) . Parts not on this list shall be submitted to GE for approval.

3.3.1.2 Selection of Materials and Processes

Selection of materials and processes shall be in accordance with GE Document (TBD) . Materials and processes not on this list shall be submitted to GE for approval.

3.3.1.3 Standard and Commercial Parts

No commercial parts shall be used without prior General Electric Company approval.

3.3.1.4 Moisture and Fungus Resistance

Wherever possible, non-nutrient materials which resist damage from moisture and fungus shall be used in the RCS design. Protective coatings shall not be acceptable as moisture and fungus preventatives for parts which may lose their coating during the normal course of assembly, inspection, maintenance and testing. The requirement of MIL-STD-454C, Requirement 4, shall apply.

3.3.1.5 Corrosion of Metal Parts

The use of dissimilar metals, as specified in MIL-STD-454L Requirement 16, shall be avoided wherever possible. Materials, techniques, and processes shall be selected and employed with regard to heat treatment procedure, corrosion protection, finish, and assembly and installation such that sustained or residual surface tensile stress, stress concentrations, and the hazards of stress corrosion, cracking, and hydrogen embrittlement are minimized. Processes and materials for protection against corrosion of metal parts shall be selected from those specified in paragraph 3.3.1.2, with the exception that cadmium plating shall not be used. Selected finishes shall be compatible with the thermal requirements of this specification. Materials and surfaces whose compatibility with hydrazine has been established shall be used for parts subjected to long term exposure to hydrazine. The subsystem shall also be internally compatible with deionized or distilled water, isopropyl alcohol as specified in TT-I-735 diluted with 2-6 percent (volume) of water, gaseous nitrogen MIL-P-27401B, air, helium MIL-P-27407

Supplement I or Argon MIL-A-18455B.

3.3.1.6 Protective Treatment

All parts shall be corrosion resistant or have a suitable corrosion resistant protective coating applied.

3.3.2 ELECTROMAGNETIC COMPATIBILITY

Electrical and electronic components of the RCS shall comply with GE Document (TBD) . Compliance with these requir ents shall be verified by test or accomplished by proof of similiarity.

3.3.3 NAMEPLATES AND PRODUCT MARKING

a. The RCS shall be marked for identification in accordance with the manufacturer's standards. The identification shall include, but not be limited to, the following:

1. Nomenclature
2. Customer Part Number
3. Serial Number (Engineering models will use a different designation than prime hardware)
4. Contract Number
5. Manufacturer's Name or Trademark
6. Date of Manufacture (month, day, year)

b. Hardware or equipment which is not suitable for use in flight, and which could be accidentally substituted for Flight or Flight Spares Hardware shall be red striped with material compatible red paint to prevent such substitution. In the event, the hardware is too small to be easily striped, or if test results would be affected by striping, a conspicuous red tag marked "NOT FOR FLIGHT USE" shall be attached.

c. Wire and Cables. Wires and cables for hardware shall not be identified by hot stamping directly onto primary or secondary (shield) insulation.

3.3.4 WORKMANSHIP

The RCS including all parts and assemblies, shall be constructed, finished and assembled in accordance with highest standards. Workmanship criteria shall comply with MIL-STD-454C Requirement 9 and 24. Particular attention shall be paid to neatness and thoroughness of soldering, wiring, marking of parts and assemblies, plating, painting, machine screw assemblage, and freedom from burrs and sharp edges. Electrical soldering shall be per requirements of NHB 5300.4 (3A).

3.3.5 INTERCHANGEABILITY

Each subassembly of the RCS and each RCS shall be directly interchangeable with regard to form, fit, and function with other subassemblies of the same part number. The requirement of MIL-STD-454C, Requirement 7, shall apply.

3.3.6 SAFETY

The RCS shall be designed to limit hazards to personnel and equipment. Explosive and toxic hazards shall be defined and procedures for limiting their effect on personnel and equipment shall be formulated and enforced. The requirements of AFWTRM-127-1 Western Test Range Safety Requirements shall apply.

3.4 MAJOR COMPONENT CHARACTERISTICS

The RCS shall consist of the following components of the type indicated. Component requirements shall be further defined to assure that subsystem performance will conform to the requirements of this specification.

3.4.1 PROPELLANT TANK ASSEMBLY

The propellant tank shall be constructed of Titanium 6AL4V, and have a minimum volume of 19,100 cu.cm. Orientation of the propellant at the tank outlet port shall be accomplished by means of either a rubber diaphragm or a surface tension device. It shall be capable of containing hydrazine (per MIL-P-26536), and shall be capable of being pressurized with gaseous nitrogen (per MIL-P-27401) to a nominal operating pressure consistent with the RCS pressure schedule. The tank shall be capable of withstanding a reverse ΔP equal to one sea-level atmosphere.

Provision for mounting the tank to the RCS structure shall be included, together with weldable (or brazeable) fittings for attachment of propellant and pressurant lines. When mounted on the RCS structure (oriented as shown on GE drawing TBD) the tank shall be capable of supplying propellant to the engines while operating under the orbital acceleration environment specified in Table III. Expulsion of propellant shall be at least 99 percent efficient.

Table III

Laterial Acceleration	(TBD) g's
Longitudinal Acceleration	(TBD) g's

3.4.2 ROCKET ENGINE ASSEMBLY

The engine assemblies shall be actuated by supplying electrical power to a normally closed propellant control valve. Each engine shall contain a combustion chamber, a catalyst bed, an expansion nozzle, a propellant injector, a propellant control valve and provision for mounting. Each engine shall employ hydrazine propellant per MIL-P-26536. A heater shall be incorporated to warm the thrust chamber catalyst bed. The propellant control valve shall provide means for direct attachment to the thruster and contain provisions for welding or brazing to the propellant line. The propellant control valve design shall permit leakage testing of each individual seat. Internal leakage of the valve seat shall not exceed 5 $s_c c$/hr GN_2 at operating pressure: The valve shall be capable of continuous power application under conditions of no-flow with no resultant damage.

3.4.3 LATCHING VALVE

The latching valve shall be electrically actuated and shall contain a latching device to maintain itself in the last energized position. The valve shall be used in the propellant line for isolation of the propellant tank assembly. The valve shall incorporate a switch for position indication, shall utilize welded or brazed inlet and outlet connections, and shall be designed for no less than 1,000 cycles from closed to open to closed.

3.4.4 FILL AND DRAIN VALVE

The fill and drain valve shall be manually operable, and shall be used for filling and draining/venting of the hydrazine or gaseous nitrogen. The valve shall provide non-interchangeable connections for pressurant and propellant usage. The valve shall contain redundant seals for external leakage and utilize welded or brazed tubing connections. The valve shall be capable of 250 operational cycles from closed to open to closed. Total external leakage of the valve, including the seat, shall not exceed 1×10^{-6} scc/sec helium at operating pressure.

3.4.5 FILTERS

A screen filter shall be supplied in the upstream portion of each thruster and latching valve. These filters shall be compatible with hydrazine and all other fluids and gases to be used in the subsystem. The maximum particle size allowed to pass shall be compatible with the thruster and latching valves.

A system filter shall be provided with filtration to a 10 micron absolute level, and this filter shall utilize welded or brazed tubing connections. Filter size shall be such that trapped dirt equal to 10 times the maximum allowed subsystem contamination will not raise the differential pressure across the filter by more than 3.5 newtons/cm^2. An etched-disc type filter is preferred.

3.4.6 PRESSURE TRANSDUCERS

An absolute pressure transducer shall be supplied in the hydrazine feed line of the subsystem. Pressure measurement tolerances shall be $\pm$ 1% full scale maximum. Pressure measurement limits shall be a minimum of 0 and 1.25 times maximum operating pressure. The pressure transducer shall not require recalibration when subjected to proof pressure.

3.4.7 PROPELLANT AND PRESSURANT MANIFOLD

Manifolding joints shall be welded or brazed. Stainless steel plumbing is preferred. Manifolds shall be designed and structurally secured to prevent excessive flexing and fatigue during vibration. No braze material shall be used which is catalytic to the propellant or to the precipitation of dissolved salts in the propellant.

3.4.8 THRUST CHAMBER HEATERS

Heaters shall be provided on all LTE thrust chambers where necessary to meet the performance requirments of this specification. Dual LTE heater elements shall be provided. Each heater shall be able to heat the thruster to and keep it at the required temperature. Nominal preheat time from 5°C to a nondegraded holding temperature shall be a maximum of 100 minutes.

3.4.9 ELECTRICAL INTERFACE PANEL

The electrical interface between the RCS and the spacecraft shall consist of a panel which contains four electrical connectors. These provide an interface between the RCS and spacecraft wiring harnesses as listed below:

a) Connector 1 - Supplies power to the solenoid valves of all LTE's

b) Connector 2 - Supplies power to and receives an output signal from all LTE temperature sensors

c) Connector 3 - Supplies power to the thrust chamber heaters of all LTE's

d) Connector 4 - Supplies power to the latching valve; supplies power to and receives an output signal from the latching valve position indicating switch, from the pressure sensor, and from the propellant tank temperature sensor.

3.4.10 WIRING HARNESS

The RCS wiring harness shall be designed per (TBD) and shall connect the spacecraft electrically with the various components of the subsystem. Where possible pigtails shall be employed at the components rather than mateable connectors. The subsystem wiring harness shall wherever possible, follow the routing of the RCS manifolding.

4.0 QUALITY ASSURANCE PROVISIONS

SECTION 9.0

SPECIFICATION

FOR

EARTH OBSERVATORY SATELLITE

POWER MODULE

TABLE OF CONTENTS

SECTION 1

SCOPE

This specification establishes the performance, design, interface, and verification requirements for the power subsystem to be used on the EOS General Purpose spacecraft segment and other earth orbiting spacecraft. For adaptability to the power requirements of different missions, the power subsystem will be modularized, with the quantity of standardized battery and electronic packages determined by the power requirements of each mission.

Solar arrays are considered to be mission peculiar equipment specifically designed for each mission and its spacecraft characteristics. Separate specification cover solar arrays in the category of mission peculiar equipment.

SECTION 2

APPLICABLE DOCUMENTS

The following documents of the exact issue shown, form a part of this specification to the extent specified herein. In the event of conflict between documents referenced here and the detail requirements in the following sections, the detail requirements of this specification shall supersede. In the event of conflict between documents referenced here and lower tier references to documents referenced here, the former supersede.

2.1 APPLICABLE DOCUMENTS

SPECIFICATIONS

National Aeronautics and Space Administration

EOS-410-02	Specifications for EOS System Definition Studies, 13 September 1974
S-311-P-11	Quality Monitoring of Integrated Circuits, 1 June 1970
S-323-P-10	Connectors, Subminiature Electrical and Coaxial Contacts for Space Flight Use, Revised December 1969

MILITARY

MIL-C-38999	Connectors, Electrical, Miniature, Quick Disconnect, Est. Reliability
MIL-C-29012A	Connectors, Coaxial, RF, General Specification for
MIL-C-26482	Connectors, Electric, Circular, Miniature, Quick Disconnect
MIL-C-17	Cables, RF, Coaxial, Dual Coaxial, Twin Conductors, Twin Head
MIL-W-81044	Wire, Electric Cross-Linked, Polyalkene, Insulated, Copper
MIL-E-5400K	Electronic Equipment, Airborne, General Specification for

General Electric

SVS XXXX	Specification for EOS General Purpose Spacecraft Segment

STANDARDS

MILITARY

MS33540C	Safety Wiring, General Practices for
MIL-STD-454B	Standard General Requirements for Electronic Equipment
MIL-STD-143A Change 1	Specification and Standards, Order of Precedence for Selection of
MS-33586A	Metal, Definition of Dissimilar
MIL-STD-130C	Identification Marking of U.S. Military Property
MIL-STD-1247A	Identification of Pipe, Hose, and Tube Lines for Aircraft, Missile and Space Systems

OTHER PUBLICATIONS

National Aeronautics & Space Administration

NHB 5300.4 (3A) May 1968	Requirements for Soldered Electrical Connections
PPL-12 Latest Issue	GSFC Preferred Parts List
NHB 5300.4 (1A)	Reliability Program Provisions for Space Systems Contractors
NHB 5300.4 (1B)	Quality Assurance Program Provisions for Space Systems Contractors

Air Force Manuals

AFM 71-4	Air Force Regulations for Transportation of Explosive and Other Dangerous Material
AFWTRM127-1	Air Force Western Test Range Safety Manual

MILITARY HANDBOOKS

MIL-HDBK-5A	Metallic Materials and Elements for Aerospace Vehicle Structure
MIL-HDBK-17	Plastics for Flight Vehicles

GENERAL ELECTRIC COMPANY

XXXXXX	EOS General Purpose Spacecraft Segment Quality Program Plan
XXXXXX	Configuration Management Plan for EOS Mission Peculiar Spacecraft Segment
XXXXXX	Reliability Program Plan, EOS Mission Peculiar Spacecraft Segment

SECTION 3

REQUIREMENTS

3.1 ITEM DEFINITION

3.1.1 ITEM DESCRIPTION

The power subsystem generates, stores, controls and distributes regulated power at 28 volts ± 1 percent for use on the EOS spacecraft. As shown on Figure 3-1 the power subsystem consists of a deployed and oriented solar array having an area of 109 sq. ft., and a Power Module which houses batteries and electronics. A slip ring assembly, not part of the Power Subsystem, transfers power and signals from the solar array to the Power Module.

A block diagram of the power subsystem is shown on Figure 3-2 indicating a direct energy transfer configuration.

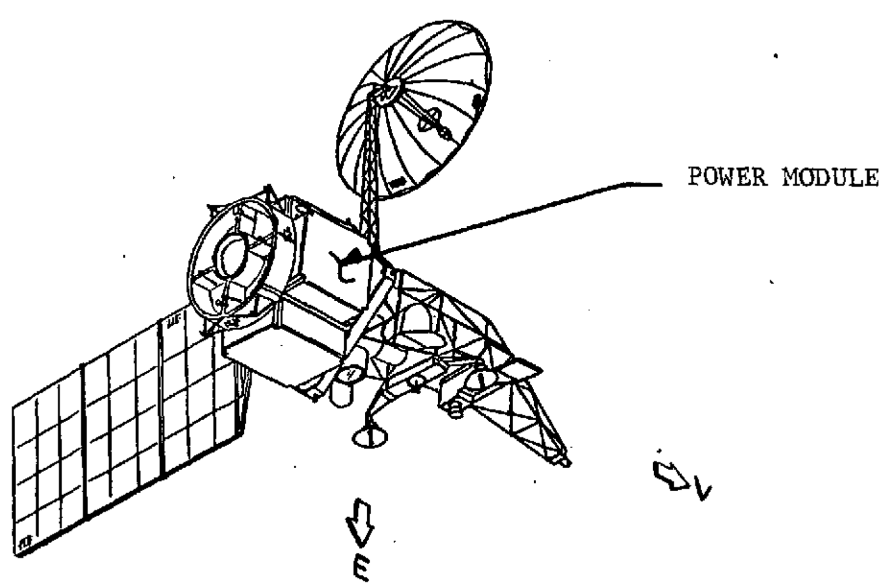


Figure 3-1. EOS Orbital Configuration

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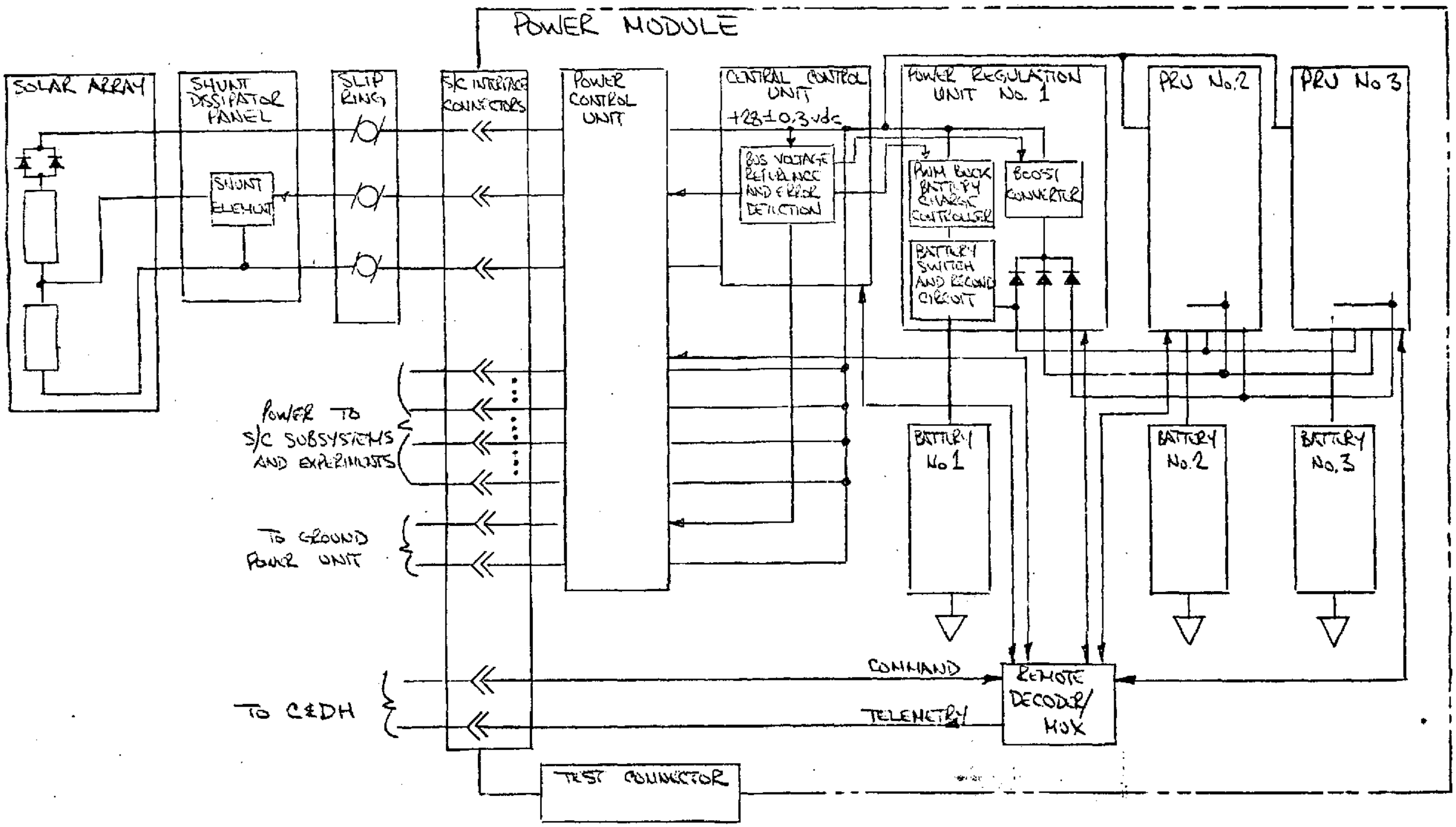

Figure 3-2. Block Diagram of Power Subsystem

Power for the solar array is supplied directly to the user loads with regulation maintained by sequential control of shunt, charge and discharge regulators by a Central Control detection circuit. Three batteries serve to provide energy storage capability. Three Power Regulation Units each contain the charge and discharge regulators for each battery. A Power Control Unit contains the power control switches, protective circuitry and other miscellaneous circuit functions. Spacecraft interface connectors provide the point of electrical connection between the power subsystem and the rest of the spacecraft. The test connector provides electrical connection to test and checkout equipment. A Remote Decoder/Mux Unit is used for command and telemetry purposes. Except for the mission peculiar solar array and Shunt Dissipator Panel, which is mounted on the solar array, all of the equipment is housed in the Power Module which has dimensions of 16 x 40 x 48 inches.

The Power Module is designed to accept additional Battery and Power Regulation Unit sets allowing power capability to be increased. Three such sets meet the EOS-A requirements; space is available for two additional sets. Such modularization can satisfy the range of anticipated earth observation missions without basic subsystem design changes.

3.1.2 INTERFACE DEFINITION

The interfaces with the power subsystem are defined below:

3.1.2.1 Electrical Interfaces

Electrical interfaces with the power subsystem are accomplished through connectors mounted on the surface of the Power Module. In general, separate connectors are used for the distinct electrical function described below.

3.1.2.1.1 Power Interface

The power subsystem supplies regulated power at 28 VDC ± 1 percent. Separate supply and return line pairs are used for the major subsystems. Power control functions (switching, overload protection) are contained in the Power Control Unit or in the user subsystem themselves depending on the function involved. Conversion to secondary voltages is performed exclusively by the user subsystem.

3.1.2.1.2 Telemetry Interface

Analog and status measurements taken within the subsystem are conditioned and routed to the Remote Decoder/Mux Unit which converts the signals to digital form and transfers them to a "party line" data bus.

3.1.2.1.3 Command Interface

Commands for the power subsystem are received by the Remote Decoder/Mux (RDM) from the "party line" data bus. The RDM interprets and routes the command signals to their proper destinations.

3.1.2.1.4 Ground Equipment Interface

The power subsystem receives ground power during prelaunch operations for operating the spacecraft subsystems and for battery charging. The power subsystem also sends signals to ground equipment for monitoring system status and receives signals from ground equipment for enabling or isolating the system. This is accomplished by direct electrical connection through the spacecraft umbilical connector. External power is similarly handled during Shuttle resupply operations.

3.1.2.1.5 Test Interface

The power subsystem provides a connector interface (besides flight connectors) for checkout of functions by test equipment.

3.1.2.2 Structural Interface

The power subsystem interfaces structurally in several segments; the Power Module, which consists of an oblong housing for the batteries and electronics, is bolt mounted to the subsystem section framework; the Solar Array is joined to the solar array drive through a bolt and flange connection; the Shunt Dissipation Panel, considered part of the solar array, is bolt mounted to the inner yoke of the Solar Array.

3.1.2.3 Thermal Interface

Heat generated within the power subsystem is primarily handled by the thermal system of the Power Module which is designed to reject heat directly to space.

3.2 CHARACTERISTICS

3.2.1 PERFORMANCE

3.2.1.1 Orbit

The power subsystem shall be designed for the following EOS-A orbit conditions:

Altitude:	418 nm
Inclination:	99°, sun synchronous
Ascending Node:	2330 o'clock

The power subsystem shall be adaptable to low earth orbits up to 900 nm and to geosynchronous orbits.

3.2.1.2 Life

The power subsystem shall provide the specified EOS power for a minimum of two years from the time of launch. Other missions will require a space operating life of 5 years.

3.2.1.3 Loads

3.2.1.3.1 Launch Loads

The power subsystem shall be capable of providing the EOS orbit average loads with battery power from five minutes before launch until solar array acquisition, a maximum of 1.75 hours after launch.

3.2.1.3.2 Initial Orbits

It may be assumed that full power orbital operation will be delayed by several orbits until the batteries are fully recharged after the initial deep discharge of the launch phase.

3.2.1.3.3 Orbit Average Loads

The power system shall be capable of supplying an EOS orbit average power of 500 watts as measured at the primary regulated bus of the power system. A preliminary breakdown of EOS orbit power demands is shown on Table 3-1 and Figure 3-3. The power system shall have a growth capability to an orbit average power of 1000 watts considering low earth orbit missions.

3.2.1.3.4 Peak Loads

The power system shall be capable of supplying a peak regulated power of 1100 watts for the EOS mission and a growth capability to 2200 watts.

3.2.1.4 Electrical Performance

3.2.1.4.1 Voltage

The primary form of distributed power shall be regulated to +28.0 $\pm$.30 volts measured at the Power Module interface connector.

3.2.1.4.2 Normal Power Transients

The disturbance voltage due to positive or negative step load changes within the rating of the system shall be within 28 $\pm$2 volts with the disturbance volt-

Table 3-1 Load Power Demand for EOS-A

Operational Mode / Subsystem	LOAD POWER DEMAND (watts)					
	Launch	Operational Average Baseload	Readout to TDRSS & LCU (6 min)	Readout to Ground Stations & LCU (3 min)	Sensor Warm-up (15 min)	Readout to LCU (3 min)
Attitude Control	91.	104.	104.	104.	104.	104.
C&DH	125.	125.	125.	125.	125.	125.
SCCM	--	84.	84.	84.	84.	84.
Reaction Control	--	20.	20.	20.	20.	20.
W/B Communications	--	--	421.	268.	--	282.
Experiments						
Data Collection System	--	40.	40.	40.	40.	40.
MSS	--	--	65.	65.	--	--
Thematic Mapper	--	10.	110.	110.	110.	110.
SUBTOTAL	216.	383.	969.	816.	483.	765.
Distribution Losses	4.	8.	19.	16.	10.	15.
Power Module	15.	15.	15.	15.	15.	15.
TOTAL	235.	406.	1003.	847.	508.	795.

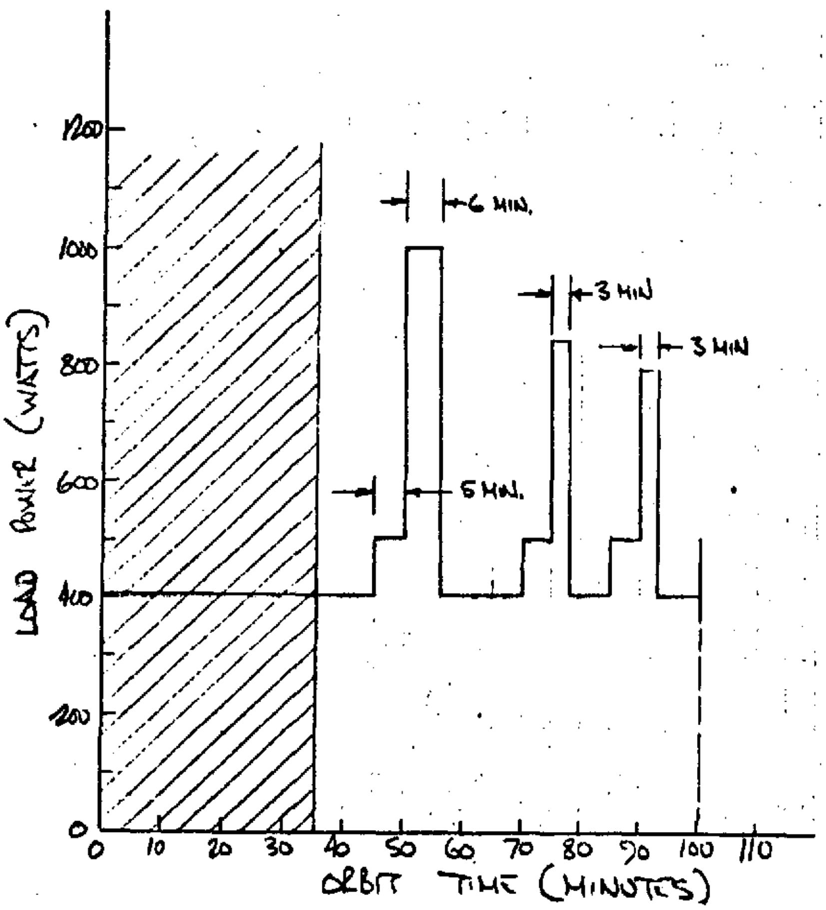

Figure 3-3 EOS-A Load Power Profile

second product not to exceed 100 microvolt-seconds. The rate of current rise or fall shall not exceed (TBD) amperes per second.

3.2.1.4.3 Power Failure Transients

Transient voltages due to a failure of a power system element shall be within limits of +18 to +33 volts with the disturbance volt-second product not to exceed 250 microvolt-seconds.

3.2.1.4.4 Ripple

Voltage ripple at the primary bus shall not exceed 100 millivolts peak-to-peak.

3.2.1.4.5 Source Impedance

The source impedance shall be
$\angle$0.15 ohms 1 Hz to 5k Hz
$<$.050 ohms 5k Hz to 100k Hz
$\angle$1.0 ohms 100k Hz to 1M Hz

3.2.2 DESIGN

3.2.2.1 <u>Electrical Interface Design</u>

Electrical interfaces with the power subsystem shall be made through module interface connectors located on the side and rear faces of the Power Module. Requirements for each interface category are specified below.

3.2.2.1.1 Solar Array Interface

Provisions shall be made on the module interface connectors for receiving solar array power through 16 pin contact pairs (16 power lines /16 returns) with each contact rated continuously at 4.5 amps under vacuum condition. Capability shall also be provided for interfacing with 12 signal circuits (24 pin contacts) from the solar array.

3.2.2.1.2 Power Distribution Interface

Provision shall be made on the module interface connectors for 10 power circuits for distributing regulated 28 volt power to the user subsystem. Each circuit shall be capable of carrying 5 amps continuously under vacuum conditions. Redundant pin contacts shall be used for the power and return legs of each circuit. Circuit breakers shall be provided for each circuit with trip settings available up to 15 amps. The circuit breakers shall be remotely resettable by command. Eight of the ten circuits shall have commandable on/off power control switches which may be combined with the circuit breaker contacts. The remaining two circuits shall be wired to user subsystem (C&DH; ACS).

3.2.2.1.3 Telemetry Interface

Telemetry outputs from the power subsystem shall be routed to the Remote Decoder/Mux supplied by the C&DH contractor and mounted in the Power Module. The telemetry output signals shall be conditioned to meet the signal requirements set forth in paragraph (TBD) of SVS (TBD).
Telemetry data requirements are listed on Table 3-2.

Two contact pairs (signal & return per pair) shall be provided for telemetry at the module interface for connection to a "party line" data bus.

3.2.2.1.4 Command Interface

Command inputs from the C&DH subsystem shall be received by the Remote Decoder/Mux Unit supplied by the C&DH contractor and mounted in the Power Module. The decoded commands will be routed to their proper destination and will be received by command circuitry designed in accordance with paragraph (TBD) of SVS (TBD). Command requirements are listed on Table 3-3.

TABLE 3-2

TELEMETRY DATA

REQUIREMENTS

FUNCTION NAME	Analog or Digital	Sample Rate (S/sec)	Origin of Signal
Battery No. 1 Charge Current	A	1/16	PRU No. 1
Battery No. 1 Discharge Current	A		PRU No. 1
Battery No. 2 Charge Current	A		PRU No. 2
Battery No. 2 Discharge Current	A		PRU No. 2
Battery No. 3 Charge Current	A		PRU No. 3
Battery No. 3 Discharge Current	A		PRU No. 3
Solar Array Current	A	1/16	PCU
Total Load Current	A	1/1	PCU
Main Bus Voltage	A	1/1	PCU
Battery No. 1 Voltage	A	1/16	PRU No. 1
Battery No. 2 Voltage	A		PRU No. 2
Battery No. 3 Voltage	A		PRU No. 3
Central Control Voltage	A		CCU
Solar Array Temperature	A		PCU
Battery No. 1 Temperature	A		PRU No. 1
Battery No. 2 Temperature	A		PRU No. 2
Battery No. 3 Temperature	A		PRU No. 3
BCC No. 1 On/Off Status	D		PRU No. 1
BCC No. 2 On/Off Status	D		PRU No. 2
BCC No. 3 On/Off Status	D		PRU No. 3
BCC No. 1 Bit 1 Status	D		PRU No. 1
BCC No. 1 Bit 2 Status	D		PRU No. 1
BCC No. 1 Bit 3 Status	D		PRU No. 1
BBC No. 2 Bit 1 Status	D		PRU No. 2
BCC No. 2 Bit 2 Status	D		PRU No. 2
BCC No. 2 Bit 3 Status	D		PRU No. 2
BCC No. 3 Bit 1 Status	D		PRU No. 3
BCC No. 3 Bit 2 Status	D		PRU No. 3
BCC No. 3 Bit 3 Status	D		PRU No. 3
Battery No. 1 Reconditioning Status	D		PRU No. 1
Battery No. 2 Reconditioning Status	D		PRU No. 2
Battery No. 3 Reconditioning Status	D		PRU No. 3
Solar Array Shunt Current	A		PCU
Battery No. 1 Discharge On/Off Status	D		PRU No. 1
Battery No. 2 Discharge On/Off Status	D		PRU No. 2
Battery No. 3 Discharge On/Off Status	D		PRU No. 3
BCC No. 1 V/T Override Status	D		PRU No. 1
BCC No. 2 V/T Override Status	D		PRU No. 2
BCC No. 3 V/T Override Status	D		PRU No. 3
All Hinge Latches Locked Status	D		PCU
Deploy/Retract Actuator Temp.	A		PCU
Solar Array Stowed Status	D		PCU
Load No. 1 On/Off Status	D		PCU
Load No. 2 On/Off Status	D		PCU
Load No. 3 On/Off Status	D		PCU
Load No. 4 On/Off Status	D		PCU
Load No. 5 On/Off Status	D		PCU
Load No. 6 On/Off Status	D		PCU
Load No. 7 On/Off Status	D		PCU
Load No. 8 On/Off Status	D	1/16	PCU

Table 3-3 Command Requirements

Command Function	Destination
BCC No. 1 ON/OFF	PRU No. 1
BCC No. 2 ON/OFF	PRU No. 2
BCC No. 3 ON/OFF	PRU No. 3
Battery No. 1 Discharge ON/OFF	PRU No. 1
Battery No. 2 Discharge ON/OFF	PRU No. 2
Battery No. 3 Discharge ON/OFF	PRU No. 3
Battery No. 1 Reconditioning ON/OFF	PRU No. 1
Battery No. 2 Reconditioning ON/OFF	PRU No. 2
Battery No. 3 Reconditioning ON/OFF	PRU No. 3
BCC No. 1 V/T Curve Bit 1	PRU No. 1
BCC No. 1 V/T Curve Bit 2	PRU No. 1
BCC No. 1 V/T Curve Bit 3	PRU No. 1
BCC No. 1 V/T Curve Bit Reset	PRU No. 1
BCC No. 2 V/T Curve Bit 1	PRU No. 2
BCC No. 2 V/T Curve Bit 2	PRU No. 2
BCC No. 2 V/T Curve Bit 3	PRU No. 2
BCC No. 2 V/T Curve Bit Reset	PRU No. 2
BCC No. 3 V/T Curve Bit 1	PRU No. 3
BCC No. 3 V/T Curve Bit 2	PRU No. 3
BCC No. 3 V/T Curve Bit 3	PRU No. 3
BCC No. 3 V/T Curve Bit Reset	PRU No. 3
BCC No. 1 V/T Override ON/OFF	PRU No. 1
BCC No. 2 V/T Override ON/OFF	PRU No. 2
BCC No. 3 V/T Override ON/OFF	PRU No. 3
Load No. 1 ON/OFF	PCU
Load No. 2 ON/OFF	PCU
Load No. 3 ON/OFF	PCU
Load No. 4 ON/OFF	PCU
Load No. 5 ON/OFF	PCU
Load No. 6 ON/OFF	PCU
Load No. 7 ON/OFF	PCU
Load No. 8 ON/OFF	PCU

Two contact pin pairs shall be provided at the module interface for receiving the command inputs from the "party line" data bus.

3.2.2.1.5 Umbilical Interface

Provisions shall be made at the module interface connector for receiving power and signals through the spacecraft umbilical connector during ground operations or Shuttle resupply operations. Capability shall be provided for transferring 20 amps through at least 4 contact pins (2 circuit pairs; ten signal circuits - 12 pins) shall also be provided for monitoring and actuation purposes.

All power and signal circuits in the Power Module which are routed through the spacecraft umbilical shall have circuit isolation provisions to prevent the effect of electrical shorts at the exposed umbilical connector. Isolation may be by incorporating the necessary isolation devices in the circuits themselves.

3.2.2.1.6 Ground Power Interface

Ground power supplied through the umbilical shall be regulated to 28 volts ± 1 percent by feedback of the Central Control voltage to the ground power supply in the same manner that the solar array is shunt regulated.

Assuming the use of a constant current shunt regulated ground power supply rated at 20 amps, the shunt regulator shall vary linearly from full-off to full-on operation corresponding to a central control feedback signal which varies from 15 to 25 volts. The feedback signal shall have a source impedance of 100K ohms.

3.2.2.1.7 Test Interface

Test connectors shall be provided on the surface of the Power Module for functionally testing the Power Module at the subsystem or system level. These connectors shall be capped in the launch configuration. The test points at the test connectors shall be resistively isolated to preclude the effects of shorting at the test connector.

3.2.2.2 Mechanical Interface

The Power Module equipment shall be housed in a container having dimensions of 16 x 40 x 48-inches. Mechanical characteristics including dimensions, mounting flange locations, handling lug size and location, and the like shall conform to the requirements of SVS (TBD).

The internal assemblies of the Power Module shall in themselves have simple bolt mounting and electrical connector interfaces. The assemblies shall be mounted to meet required thermal and environmental conditions. They shall also be mounted to permit the removal of any single assembly without disturbing the mechanical and electrical connection to other assemblies.

A single Remote Decoder/Mux Unit shall be housed in the Power Module. Its installation characteristics shall be furnished by the C&DH contractor.

3.2.2.3 Thermal Interface

The batteries of the Power Module shall be located and mounted so as to maintain their temperatures within limits of 0 to 20°C. Other Power Module equipment shall be maintained within temperature limits of (TBD) to (TBD) °C.

3.2.2.4 Solar Array Design

Solar array design requirements are specified in mission peculiar specifications. EOS-A solar array requirements are specified in SVS (TBD).

3.2.2.5 Power Module Design

3.2.2.5.1 Weight

The weight of the Power Module shall not exceed 292 lbs. The weight breakdown shall be as follows:

Quantity	Item	Total Weight (lbs)
1	Central Control Unit	4
3	Power Regulation Unit	45
3	Batteries	141
1	Power Control Unit	30
1	Remote Decoder/Mux	2
	Harness	30
	Housing	40
		292

3.2.2.5.2 Central Control Unit

The Central Control Unit shall provide driver signals proportional to the bus voltage deviation from a 28 volt reference level for operating the solar array shunt regulator, the charge and discharge regulators in the Power Module, and the ground shunt-regulated power supply. The driver signals shall be of sufficient magnitude for operating 2 to 5 sets of charge/discharge regulators, the shunt regulators for handling a 70 amp 28 volt output solar array, and varying the output of a 20 amp 28 volt ground power supply.

The Central Control Unit shall be designed so that no single piece part failure results in any loss of function.

3.2.2.5.3 Batteries

Each battery shall contain 17 series-connected 20 amp-hour sealed nickel-cadmium cells. The EOS-A spacecraft shall utilize three batteries. The Power Module shall have sufficient room for five batteries.

Three thermistors shall be mounted on each battery for monitoring temperatures for telemetry, charge control and charge cutoff purposes.

Power and signal connections to the battery shall be made through electrical connectors. A special test connector shall be installed for checking cell voltages.

3.2.2.5.4 Power Regulation Unit (PRU)

The PRU shall contain charge and discharge regulators for each battery. The PRU's shall be mounted in the Power Module with room for five units. Three units shall be used on EOS.

The PRU shall perform the following functions:

a. Receive and supply power at +28 volts.

b. Receive driver signals from the Central Control Unit for modulating power delivered by the boost regulator and inhibiting battery charge.

c. Control battery charging by means of a Charge Regulator. The Charge Regulator shall have a 7 amp current limit until the battery voltage reaches one of the selected battery temperature compensated voltage limit curves shown on Figure 3-4. The current shall then be reduced to limit the voltage in accordance with the selected curve. Charge current shall be completely interrupted for battery temperature equal to or greater than 35°C. Charge current shall also be reduced linearly in response to an inhibit driver signal from the Central Control. The pass element shall be implemented using a buck PWM circuit. The Charge Regulator efficiency shall not be less than 90 percent. It should be possible to turn off the charge regulator by command (zero charge current).

C-4

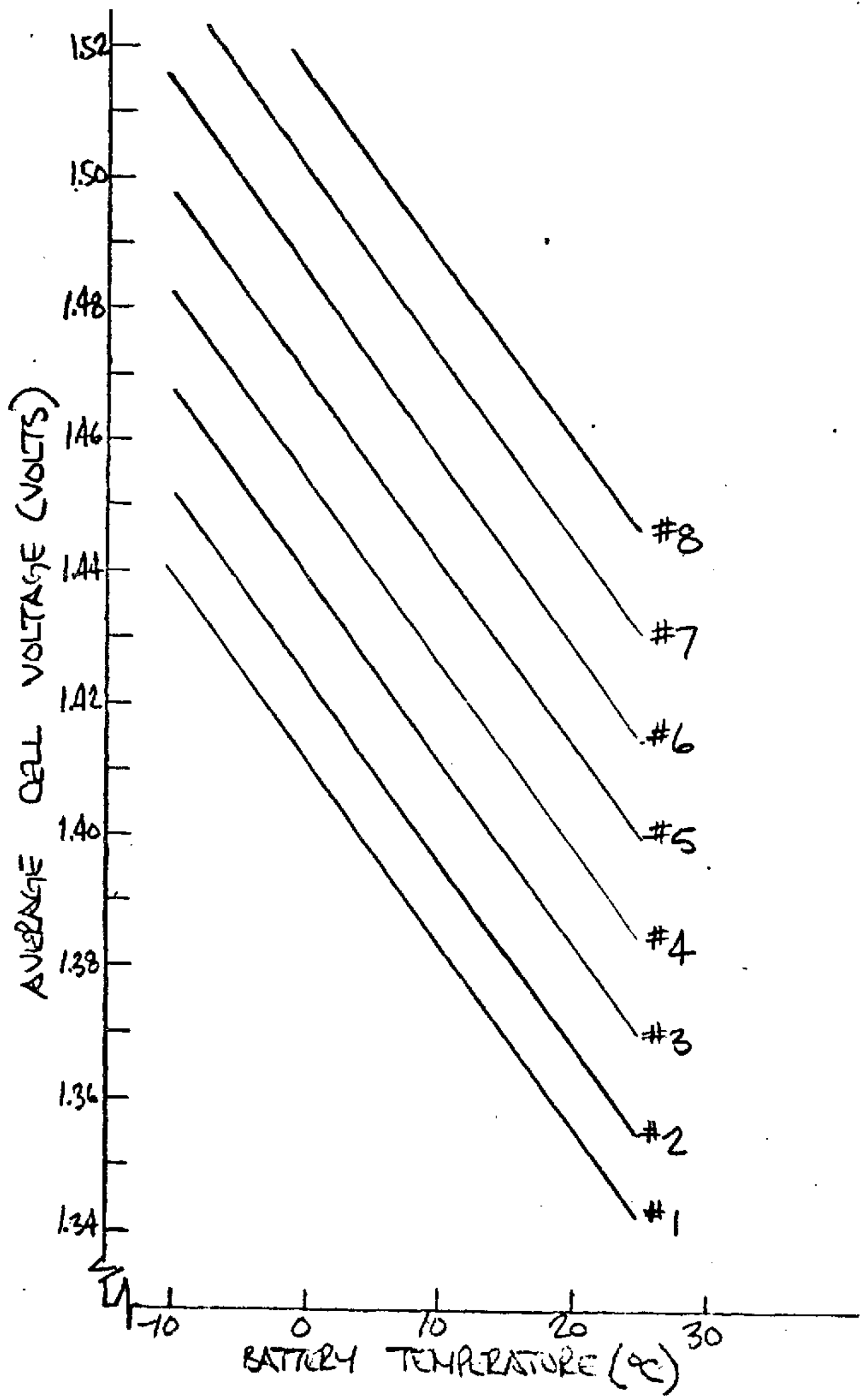

Figure 3-4 Typical Battery Temperature Compensated Voltage Limits

d. Control battery discharge by means of a Boost Regulator. The Boost Regulator shall supply power to the regulated bus in response to the magnitude of a driver signal from the Central Control. The unit shall have a power capability of 450 watts with a corresponding efficiency of no less than 90 percent. At a power transfer of 150 watts the unit shall have an efficiency of no less than 92 percent. Intermediate efficiencies shall be linearly related to these ratings. The unit shall have a current limit of 16 amps. Standby losses shall not exceed 6 watts. A boost PWM circuit implementation shall be used with circuit redundancy preventing loss of function with any single piece part failure.

e. Provide redundant battery discharge isolation diodes. The point between the diode and the input to the boost regulator of all PRU's shall be tied in common so that all batteries can provide input to any single boost regulator.

f. Provide an isolation switch in the positive line to the battery. The switch shall be of the latching relay type operable by common or directly through umbilical wiring from ground or Shuttle equipment. In the latter case power for operating the relay shall be derived from the remote ground or Shuttle power sources.

g. Provide telemetry outputs to the Remote Decoder/Mux. The telemetry signals shall be conditioned to watch the RDM input requirements in accordance with paragraph (TBD) of SVS (TBD). The telemetry measurements pertain to the performance of the specific PRU and its associated battery and shall be as follows:

Measurement	Range
Battery voltage	(later)
Battery charge current	
Battery discharge current	
Battery temperature	
Battery isolation switch on/off status	
Selected charger voltage limit	(1 of 8 positions)
Charger on/off status	

h. Receive command inputs from the Remote Decoder/Mux. Receiving circuitry shall watch the requirements set forth in paragraph (TBD) of SVS (TBD). Commandable functions for each PRU shall be as follows:

Command Function	No. of Commands
Turn charger on/off	2
Select charger voltage limit	4
Battery isolation switch on/off	2

The PRU shall be bolt-mounted to the Power Module with provision for conducting away internally generated heat. All electrical connections shall be made through pin type connectors.

3.2.2.5.5. Power Control Unit (PCU)

The PCU shall contain power control switches, protection devices and miscellaneous circuitry. The PCU shall perform the following functions:

a. Receive power from the solar array; transfer power to and receive power from each PRU/battery set; distribute power through ten distribution circuits; receive power from ground or Shuttle power sources.

b. Provide fault protection for the ten distribution circuits. Protection shall be in the form of command resettable circuit breakers with selectable trip ratings up to 15 amps.

c. Provide power control switches for eight of the ten power distribution circuits. Switches shall be of the latching type with remote command on/off inputs.

d. Provide two isolation switches for the input solar array power and the main bus ahead of the ten distribution circuits. These switches shall be hardline actuated from the ground or Shuttle through the spacecraft umbilical. Switch actuator power shall be derived from 28 volt ground or Shuttle power sources.

e. Perform telemetry measurements and condition signals for transmittal to the RDM's in accordance with paragraph (TBD) of SVS (TBD) . The telemetry measurements shall be as follows:

Measurement	Range
Main bus voltage	later
Solar array current	
Main bus current	
Solar array temperature*	
Solar array shunt current*	
On/off status of power control switches	

* signals received from solar array and conditioned in PCU.

f. Receive command inputs.

3.2.2.5.6 Remote Decoder/Mux (RDM)

The RDM shall be supplied by the C&DH contractor and shall be designed in accordance with SVS (TBD).

3.2.2.5.7 Harness

The harness shall electrically interconnect the equipment housed in the Power Module. The harness shall be modular design for maximum system flexibility. As PRU's and batteries are added, they shall be electrically interconnected to the system by adding harness segments. The harness shall be designed to permit easy removal of assemblies with minimal disturbance to the harness and other assemblies.

Separate connectors shall be used to segregate power and signal lines.

Multiple connectors for similar functions shall be used on the Power Control Unit, the Central Control Unit and the Remote Decoder/Mux to accommodate the addition of Power Regulation Units and Batteries to minimize harness redesign.

Cable strain relief devices or backshell potting shall be employed at all harness terminations.

Wire sizes shall be selected to hold round trip voltage drops between source and load to one percent or less of the supply voltage, except that loads of greater than 100 watts may have round trip voltage drops as high as 500 mv. The minimum wire size for power and control circuitry shall be AWG #20. The minimum wire size for data or test circuitry shall be AWG #22. Under worst case conditions, wire temperature shall not exceed the temperature rating of the wire insulation.

Particular attention should be given to the harnessing with regard to wire type, current derating, wire and contact size for a given function, isolation between wires for certain functions, bend radii, methods of termination (including shields), the types of connectors used, stress relief at wire terminations etc. For this system, the use of crimped contacts is preferred. In addition, the use of connectors with bayonet or threaded couplings are preferred over other types of connector engagement methods.

3.2.2.5.8 Electrical Design

A. Grounding Configuration. The power subsystem shall be designed to interface properly by implementinn of the following grounding configuration in the power subsystem module:

1). Chassis Grounds. All chassis within the module shall be electrically bonded to the module. The module will be electrically bonded to the spacecraft structure via the module/structure interface connector and mechanical interfaces.

2). Signal Grounds. All signal grounds within the module shall be returned to a common point within the power module. This point shall be routed to a central ground point on the spacecraft structure via the module/structure interfaces connector. Signal ground shall be isolated from power ground and chassis ground within the module except for EMI or audio frequency bypass capacitors between signal ground and chassis ground and between power ground and chassis ground.

3). Power Grounds. All power grounds (returns) from within the power module and from the array(s) and spacecraft loads shall be returned to the central ground point via the module/structure interface connector. Power ground shall be isolated from signal ground by the use of converters or inverters except for the common connection at the central ground point on the spacecraft structure. Heaters shall be powered from the DC source and shall be returned to power ground. EMI filter capacitors shall be grounded directly to the chassis or module structure.

4). Shield Grounds. All shields shall be grounded to chassis or structure at each end.

B. Electromagnetic Compatibility. The power subsystem module shall be designed to minimize the radiation of self generated noise and shall be shielded to preclude the possibility of susceptibility to EMI from spacecraft or external sources. For specific missions it shall be possible to incorporate additional shielding to further reduce radiation or susceptibility. System design shall be based on suppression of noise at its source and the containment of self generated noise within the generating assembly. Good design practices in chassis design, EMC filtering, grounding, bonding, etc. shall be employed.

SECTION 10.0

Specification SVS XXXX
16 September 1974

SPECIFICATION

FOR THE

EARTH OBSERVATORY SATELLITE (EOS)

ELECTRICAL INTEGRATION

SUBSYSTEM

TABLE OF CONTENTS

1.0 SCOPE

This specification establishes the performance, design, and interface requirements for the electrical integration subsystem to be used on the EOS-A spacecraft. The techniques defined provide the flexibility to service all the missions defined for the EOS spacecraft; however, some design changes are necessary to satisfy the unique requirements of each mission. The biggest impact occurs in the design of the Signal Conditioning and Control Module (SCCM) which provides circuitry to service mission unique functions such as solar array deployment and drive, antenna deployment, and shuttle signal conditioning. Standard signal distribution such as command, telemetry, and timecode is handled with data busses which meet varying mission requirements without design change. Distribution of other signals (power, standard clock) is provided on an as needed basis from a universal interface.

2.0 APPLICABLE DOCUMENTS

The following documents of the exact issue shown from a part of this specification to the extent referenced herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

2.1 Applicable Documents

SPECIFICATIONS

National Aeronautics and Space Administration

EOS-410-02	Specifications for EOS System Definition Studies, 13 September 1974
S-311-P-11	Quality Monitoring of Integrated Circuits, 1 June 1970
S-323-P-10	Connectors, Subminiature Electrical and Coxial Contacts for Space Flight Use, Revised December 1969

MILITARY

MIL-C-38999	Connectors, Electrical, Miniature, Quick Disconnect, Est. Reliability
MIL-C-29012A	Connectors, Coaxial, RF, General Specification for
MIL-C-26482	Connectors, Electric, Circular, Miniature, Quick Disconnect
MIL-C-17	Cables, RF, Coaxial, Dual Coaxial, Twin Conductors, Twin Head
MIL-W-81044	Wire, Electric Cross-Linked, Polyalkene, Insulated, Copper
MIL-E-5400K	Electronic Equipment, Airborne, General Specification for

GENERAL ELECTRIC

SVS XXXX	Specification for EOS General Purpose Spacecraft Segment
SVS XXXX	Specification for EOS Mission Peculiar Spacecraft Segment

MILITARY

MS33540C	Safety Wiring, General Practices for
MIL-STD-454B	Standard General Requirements for Electronic Equipment
MIL-STD-143A Change 1	Specification and Standards, Order of Precedence for selection of
MS-33586A	Metal, Definition of Dissimilar
MIL-STD-130C	Identification Marking of US Military Property
MIL-STD-1247A	Identification of Pipe, Hose, and Tube Lines for Aircraft, Missile and Space System

OTHER PUBLICATIONS

National Aeronautics and Space Administration

NHB 5300.4 (3A) May 1968	Requirements for Soldered Electrical Connections
PPL-12 Latest Issue	GSFC Preferred Parts List
NHB 5300.4 (1A)	Reliability Program Provisions for Space Systems Contractors
NHB 5300.4 (1B)	Quality Assurance Program Provisions for Space Systems Contractors

Air Force Manuals

AFM 71-4	Air Force Regulations for Transportation of Explosive and Other Dangerous Material
AFWTRM127-1	Air Force Western Test Range Safety Manual

MILITARY HANDBOOKS

MIL-HDBK-5A	Metallic Materials and Elements for Aerospace Vehicle Structure
MIL-HDBK-17	Plastics for Flight Vehicles

INTERSTATE COMMERCE COMMISSION

T.C. George's Tariff No. 6C	Transportation of Explosives and Other Dangerous Articles by Commercial Aircraft
T.C. George's Tariff No. 19	Transportation of Explosives and Other Dangerous Articles by Land, Water, in Rail Freight Service and by Motor Vehicle (Highway) and water.

GENERAL ELECTRIC COMPANY

XXXXX	EOS General Purpose Spacecraft Quality Program Plan
XXXX	Configuration Management Plan for EOS General Purpose Spacecraft
XXXX	Reliability Program Plan, EOS General Purpose

3.0 REQUIREMENTS

3.1 DESCRIPTION

The electrical system for the EOS spacecraft provides the signal distribution among the modules comprising the basic spacecraft bus and the mission peculiar equipment. It also includes the Signal Conditioning and Control Module (SCCM) which contains circuitry necessary for spacecraft operation, but not provided by the three standard spacecraft modules (ACS, Power, C&DH). A block diagram of the spacecraft electrical system is given in Figure 3-1A and 3-1B. This diagram shows the three basic modules, the SCCM, the propulsion module and the omnidirectional S-Band antenna, which comprise the basic spacecraft bus. It also shows the EOS-A mission peculiar equipment consisting of the solar array, wideband module, thematic mapper, MSS,TDRSS antenna, and STDN antennas. The electrical system provides signal distribution between the basic spacecraft bus and the mission peculiar equipment via an interface panel in the transition frame, except for the mission peculiar signals which originate in the SCCM. The electrical system also provides the unbilical interface which is used for signal distribution to and from the launch vehicle, ground servicing equipment, or shuttle.

The power module provides +28 $\pm$ 0.3VDC regulated bus to each of the spacecraft subsystems. Separately buffered outputs are provided to each user as his primary power source. An additional output is provided as a heater power source and is common to all modules. All outputs monitor current and are protected against overload. Each output, except the ACS and C&DH module outputs, are capable of being switched by ground or OBC command. The power module also interfaces with the umbilical connector for battery conditioning capability.

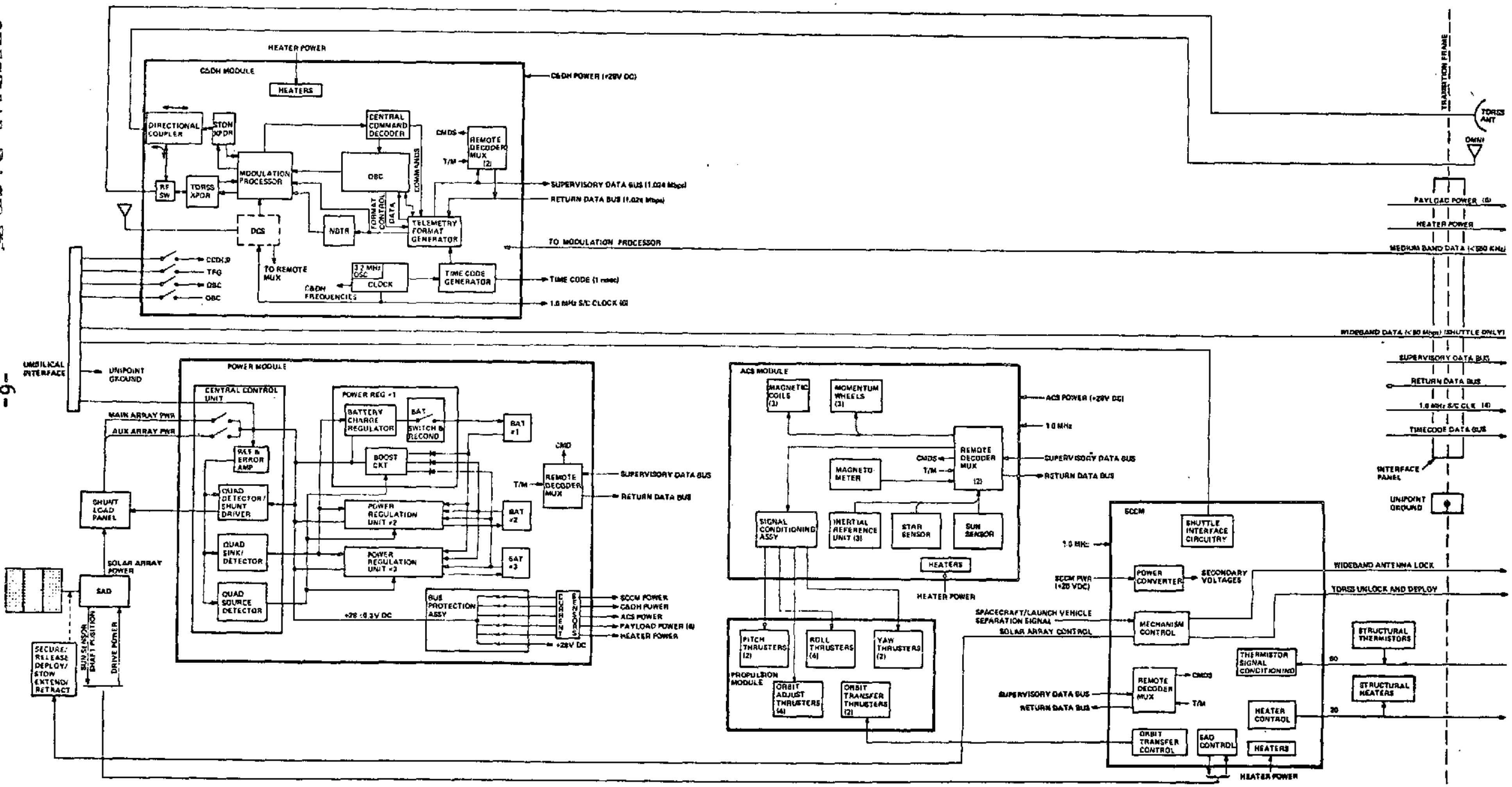

Figure 3.1A. General Purpose Spacecraft Segment
Spacecraft Electrical System Block Diagram

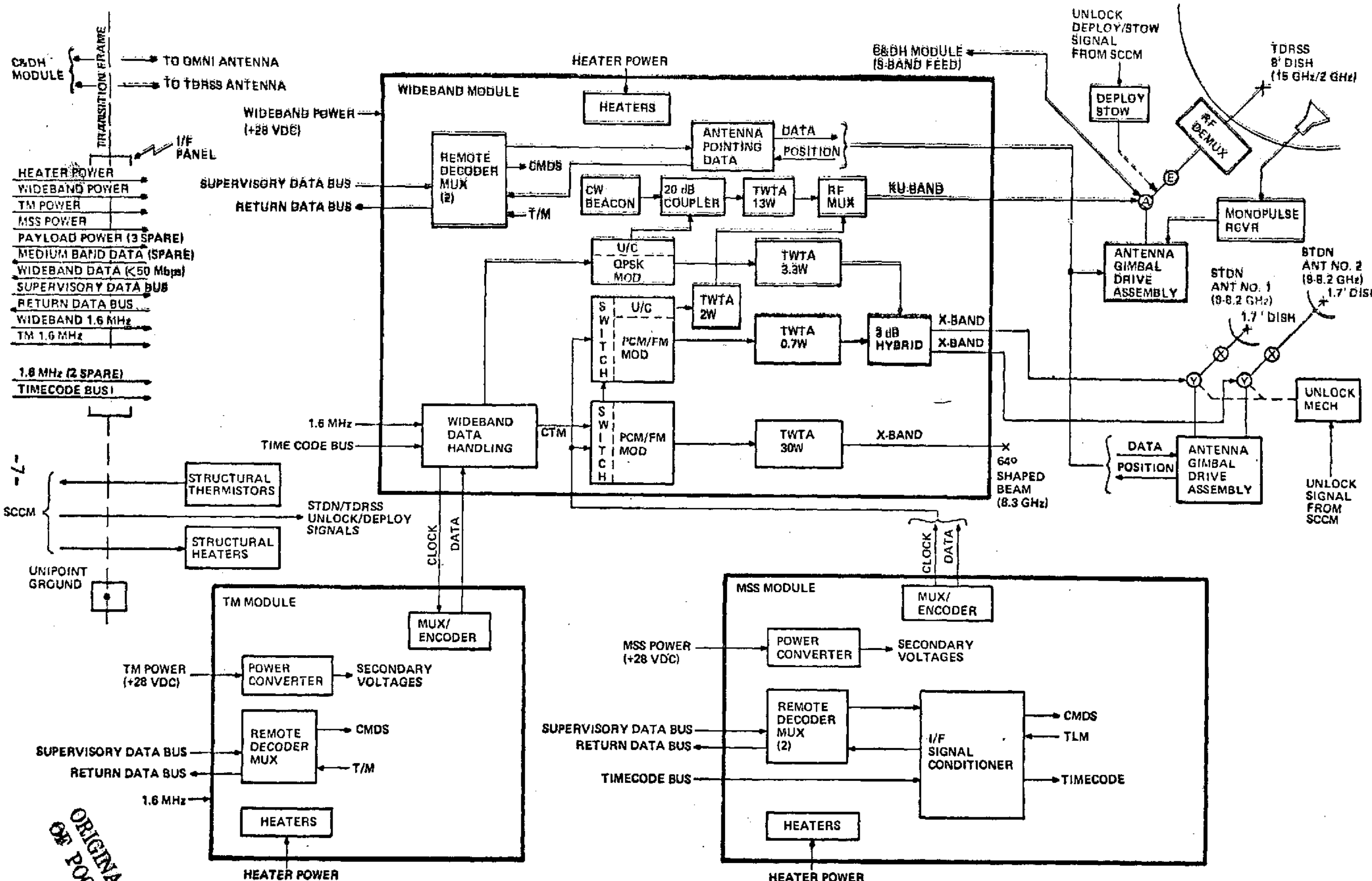

Figure 3.1B. Mission Peculiar Spacecraft Segment System S/C Block Diagram

The C&DH module provides spacecraft tracking, on board control of all spacecraft and payload functions, and retrieval of narrowband and mediumband ($<$ 650 Hz) observatory data. Commanding is accomplished via a supervisory data bus which is capable of supporting 32 remote decoder/muxes located in the user subsystems. Capability exists for executing 2048 pulse commands and 128 serial magnitude (16 bit) commands from the ground or OBC. Telemetry data are analog, bilevel digital, or serial digital. A total of 2048 different functions can be handled, with a maximum of 512 of these being serial digital. Data are formatted into a 128 x n (n is a binary number $\leq$ 128) word major frame consisting of 124 columns of sampled data (including four subcommutated columns). Synchronization and minor frame ID data are inserted into the remaining four columns. Timecode data are inserted into the first row of each major frame in the four subcommutated columns. The C&DH module also provides separately buffered standard 1.6 MHz clock to the spacecraft subsystems. Timecode data are distributed on a common data bus. Mediumband data ($<$ 650 kHz) are accepted for application to the STDN downlink in lieu of GRARR. The C&DH module is also capable of housing a data collection system and antenna. C&DH ground communication is accomplished through either an omnidirectional antenna or the S-band feed of the TDRSS antenna. The C&DH module also interfaces with the umbilical for shuttle command and telemetry control.

The AGC module provides the sensors and reaction control devices necessary for providing proper spacecraft attitude based on computations performed by the OBC in the C&DH module. It also provides the valve drivers which interface with the reaction control and orbit adjust thrusters located in the propulsion module. The ACS/OBC communication link is provided by the command and telemetry supervisory and return data busses.

The SCCM contains the circuits necessary for controlling spacecraft functions not in the standard modules. These include structure heater control, structure thermistor signal conditioning, solar array drive control, adapter separation, solenoid drivers, pyro drivers, appendage deployment, orbit transfer control, and shuttle interface. It also provides an interface with the umbilical for providing launch vehicle, ground support, and shuttle communications.

The propulsion module houses the thrusters necessary for reaction control, orbit adjust, and orbit transfer. Driver circuitry for these functions is located in either the ACS module or the SCCM.

The wideband module obtains digital data from the thematic mapper and MSS and provides data annotation, conditioning and compaction as necessary. It also provides the switching and transmission equipment necessary for the wideband communications downlink. The downlink data is transmitted via the TDRSS Ku-band feed, the STDN dishes, and the low cost user shaped beam antenna. The servo control circuits of the gimballed TDRSS and STDN antennas are housed in the wideband module with one loop control date provided by the OBC based on algorithmic updates.
The Thematic Mapper and the MSS form the payload complement of EOS-A.

3.2 CHARACTERISTICS

3.2.1 PERFORMANCE

3.2.1.1 Signal Conditioning and Control Module (SCCM)

The Signal Conditioning and Control Module (SCCM) shall contain circuitry for performing the following functions:

Solar Array Drive Control

Orbit Transfer Control (solenoid activation)

Structural Heater Control

Structural Thermistor Signal Conditioning

Mechanism Control (solar array deployment, wideband antenna release, TDRSS antenna deployment)

Shuttle Interface Circuitry

The circuitry requirements associated with each of these functions are defined below.

A. SOLAR DRIVE ARRAY CONTROL

1. Solar array shall point to sun within ± 5 degrees during the daylight portion of each orbit.
2. Stepper motor duty cycle shall be as close to continuous as possible to minimize perturbations of the ACS.
3. Drive shall be capable of being stopped or operated at a slew rate (8 x normal rate).
4. Stepper motor drives shall be redundant.

B. ORBIT TRANSFER CONTROL

1. Firing circuits shall be safed; i.e., more than a single command shall be required to perform the function. Arming function shall activate a timer which shall disable the arming circuit after ten seconds. Fire commands shall be issued within this ten-second period. Thruster disable commands shall be accepted at any time.

2. Arming and firing circuits shall be redundant.

C. STRUCTURAL HEATER CONTROL

Twenty separate heater control circuits shall be provided. Each heater control circuit shall be capable of handling up to five watts.

D. STRUCTURAL THERMISTOR SIGNAL CONDITIONING

Sixty thermistor signal conditioning circuits shall be provided.

E. SOLAR ARRAY DEPLOYMENT

1. Provision shall be made to effect deployment of the solar array TBD seconds after the spacecraft separates from the adapter. An adapter separation signal will be provided by means of a switch in the spacecraft which will close upon adapter separation. The signal will be transferred to the SCCM to begin the deployment sequence.

2. Solar array deployment shall occur in three steps: release, deploy, and extend. Capability for retract shall also be provided. (Stow and secur functions, if necessary, will be performed by shuttle).

3. Operating circuits shall be safed such that none of the functions can occur prior to adapter separation.

F. TDRSS ANTENNA DEPLOYMENT

1. TDRSS antenna deployment shall occur in two steps: release and deploy. Capability for retract shall also be provided. (Stow and secure functions, if necessary will be performed by shuttle).

2. Operating circuits shall be safed such that none of the functions can occur prior to adapter separation.

G. STDN ANTENNA RELEASE

1. Operating circuits shall be safed; i.e., more than a single command shall be required to perform function.
2. Arming and firing circuits shall be redundant.

H. SHUTTLE INTERFACE CIRCUITRY (VIA UMBILICAL)

1. Capability shall be provided to disable the spacecraft power bus from the solar array and provide input power from a solar array simulator in the shuttle.
2. Caution and warning signals shall be made available to the shuttle directly from monitors, in addition to telemetry matrix data.
3. Direct command and control capability shall be made available to shuttle to enable response to caution and warning signals.
4. Safing of all pyro circuits shall be provided through disabling of arming circuits using holding relays activated by shuttle.

3.2.1.2 Signal Distribution

The separately buffered regulated bus outputs shall be distributed to the spacecraft subsystems as follows: one each to the C&DH, ACS, SCCM and six to the interface panel on the transition frame (three of these shall be distributed to the wideband module, the thematic mapper, and the MSS; the other three shall be spare). The heater bus output from the power module shall be distributed to the C&DH, ACS, SCCM, and the interface panel for subsequent distribution to the wideband module, thematic mapper, and MSS. A maximum of 280 mv line drop to loads requiring 100 watts or less and 500 mv line drop to greater loads shall be maintained throughout the power harness. All power return lines shall be referenced to a power ground within the power module which is maintained at equipotential with the spacecraft unipoint ground located on the spacecraft transition frame. Provision shall also be made for receiving battery conditioning power through the spacecraft umbilical connector from ground support equipment or shuttle. Capability shall be provided for transferring 20 amperes through at least four contact pins. These inputs shall be isolated against shorts in the umbilical when not in use.

The C&DH module supervisory and return data busses and the timecode data bus shall be transformer coupled at the C&DH module and each remote. The lines shall be balanced and not tied to any ground. The driving and receiving logic shall be referenced to local signal ground which is tied to the spacecraft unipoint ground. The supervisory and return data busses shall be tied to remote decoder/muxes in the power module, C&DH module (2 internal), ACS module, SCCM (2), wideband module (2), thematic mapper and MSS (2). Distribution to the wideband module, thematic mapper and MSS shall be made through the transition frame interface panel. Timecode distribution shall be made to the wideband module and MSS through the transition frame interface panel. The 1.6 MHz standard clock output shall be distributed from six separately buffered balanced outputs. One shall be provided to the ACS, one to the SCCM, and four to the transition frame interface panel (one is subsequently routed to the wideband module and one to the thematic mapper; two are spare). A distribution harness shall also be provided from the C&DH module to the spacecraft unbilical for command and telemetry access. RF connections shall be provided to the omnidirectional-antenna and the S-band feed of the TDRSS antenna. A third RF interface shall be provided for a possible link between an external UHF antenna and an internal data collection system. One additional signal interface shall be provided for possible use by a medium band sensor. This interface shall be made through the transition frame interface panel.

Distribution of all signals into and out of the signal conditioning module shall be made by direct connection between the source and user on an as needed basis. This includes all actuator, drive, heater control, and sensor signals.

Distribution harnesses shall be provided between the wideband module and the thematic mapper and MSS for collection of high rate digital data ($>$20 Mbps).

Distribution of RF signals from the wideband module shall be made to each of four antennas: TDRSS (Ku-band feed), STDN (2), and LCU. The TDRSS and STDN signals shall be distributed via waveguides in the rotary joints of their respective gimbal assemblies.

A control and feedback signal harness shall also be provided between the wideband module and the three antenna gimbal assemblies and between the monopulse receiver located on the TDRSS dish and its gimbal assembly.

3.2.1.3 Transition Frame Interface Panel

Table 3-1 lists the functions which shall be provided on the transition interface panel.

3.2.1.4 Umbilical Interface

Table 3-2 lists the functions which shall be provided on the umbilical interface connector.

3.2.2 DESIGN

3.2.2.1 Electrical

Table 3-1. Transition Frame I/F Panel

Signal	No. of Pins	Cable Type
Module Power (Redundant)	12 + 12 RTN	T2
Heater Power (Redundant)	2 + 2 RTN	T2
Supervisory Data Bus (Redundant)	2 + 2 RTN	T2S
Return Data Bus (Redundant)	2 + 2 RTN	T2S
Timecode (Redundant)	2 + 2 RTN	T2S
1.6 MHz Clock	4 + 4 RTN	Twin-ax
Mediumband Data	1	SCS
Wideband Data	1	Coax
RF Signal (Spare)	2	Coax
Spare	10	--

Table 3.2. Umbilical I/F

Signal	No. of Pins	Cable Type
Battery Conditioning	2 + 2 RTN	T2
Command Input (Digital)	1 + 1 RTN	T2S
Data Output (Digital)	1 + 1 RTN	T2S
OBC I/F	2 + 2 RTN	T2S
C & W (Shuttle)	25	SCS
Command Control	10 + 10 RTN	T2S
Unipoint Ground	5	SC

3.2.2.1.1 Signal Conditioning and Control Module. The following interface requirements shall be provided by the EOS-A SCCM:

A. SOLAR ARRAY DRIVE CONTROL

1. Redundant +28 VDC output pulses (~20 μsec duration) to drive each SAD stepper motor (windings = 80 ohms). These pulses shall be derived from the 1.6 Mhz input clock and (when in the NORMAL mode) be capable of driving the motor at or slightly faster than orbital rate (TBD). The motor requires 334.5 pulses per degree of movement. A rate shall be chosen such that the paddles shall not be driven out of the ±5 degree reference to the sun with continuous operation during the daylight portion of the orbit. The motor shall be driven at the same rate during the night portion of the orbit. The pulse train shall be interrupted when the sun sensor indicates the paddles are more than 5 degrees ahead of the sun and restored when the sun sensor indicates the paddles are 5 degrees behind the sun. The SLEW mode shall drive the motor continuously at 8 x NORMAL rate and not respond to sun sensor inputs. An acquisition mode shall be defined to put the drive in the SLEW mode upon receipt of the paddle extend signal until the sun sensor indicates a null. The drive shall then be switched to NORMAL mode.
2. Sun sensor and paddle position inputs shall be conditioned to 0 to +5 VDC with a source impedance of 5 kohms, maximum and applied to the return data bus when addressed by the TFG (once per major frame). The sun sensor output shall be an analog signal and the paddle position output a serial digital word consisting of the 8 MSB of an 11-bit counter which is reset to zero by the reference pulse.

3. Digital status of the mode of operation (ON/OFF, NORMAL/SLEW), shall be conditioned to digital "0" = 0.0 $\pm$0.5 VDC, digital "1" = +5 $\pm$0.5 VDC, source impedance 5 kohms, maximum, and applied to the return data bus when addressed by the TFG (once per major frame).

B. ORBIT TRANSFER

1. Six redundant pairs of solenoid valve driver pulses with the following characteristics: 28 VDC, 10 amps, 100 msec.

2. Digital indication of the firing bus status (BUS 1 ARM/DISARM, BUS 2 ARM/DISARM) shall be conditioned and applied to the return data bus when addressed by the TFG (twice per major frame).

C. STRUCTURAL HEATER CONTROL

1. Twenty separately fused and switched outputs of regulated bus to heaters mounted on the spacecraft structure. Each circuit shall be capable of supplying five watts continuously.

2. Digital status of each of the heater circuits (HEATERS ON/OFF) shall be conditioned and applied to the return data bus when addressed by the TFG (once per major frame).

D. STRUCTURAL THERMISTOR SIGNAL CONDITIONING

1. Biasing voltage to each thermistor. Each output shall be current limited such that a short will not draw more than 5 milliamps.

2. Conditioned analog telemetry for each thermistor shall be applied to the return data bus when addressed by the TFG (once per major frame).

E. ADAPTER SEPARATION SIGNALS

1. +28 VDC regulator bus to one side of each of two microswitches in the adapter.

2. Redundant separation signals with the following characteristics: TBD. Each of these signals shall be issued from one of two timers. The first shall occur TBD seconds after adapter separation.

The second shall occur 5 seconds after the initial timer, but shall not be dependent on the initial timer.

3. Digital status of each of the adapter separation signals (SEPARATION YES/NO) shall be conditioned and applied to the return data bus when addressed by the TFG (once per major frame).

F. SOLAR ARRAY DEPLOYMENT

1. Seven redundant pairs of solenoid driver pulses (to paddle unlatch and hinge pins) with the following characteristics: +28 VDC, 10 amps, 100 msec. Paddle unlatch shall be activated by adapter separation signal (s).

2. +28 VDC output pulses derived from the 1.6 MHz clock (rate, voltage, and pulsewidth TBD) to drive paddle deploy stepper motor windings. These pulses shall be activated by the paddle unlatch signal and interrupted with the fully deployed and locked signal.

3. +28 VDC output pulses (rate, voltage, and pulsewidth TBD) derived from the 1.6 MHz clock to drive paddle extend stepper motor windings. Pulse sequence to windings shall be reversible in order to effect paddle retract. Extend pulses shall be enabled by command, activated by the fully deployed and locked signal, and interrupted with the fully extended signal. Retraction shall be by command (after unlocking hinge pins) by reversing application of pulses to the motor windings and interrupted by the fully retracted signal.

4. +28 VDC biasing voltage for the unlatch, hinge pin, extend, and retract indicators.

5. Digital indication of the mechanism status (ARM/DISARM, LATCH/UNLATCH, HINGE PIN LOCK/UNLOCK,(3), EXTEND COMPLETE YES/NO, RETRACT COMPLETE YES/NO) shall be conditioned and applied to the return data bus when addressed by the TFG (once per major frame).

G. TDRSS ANTENNA DEPLOYMENT

1. Design of this circuit shall be similar to that used for solar array deployment.

H. STDN ANTENNA RELEASE

1. Two redundant pairs of solenoid driver pulses with the following characteristics: +28 VDC, 10 amps, 100 msec.
2. Digital indication of the firing bus status (ARM/DISARM (2)) shall be conditioned and applied to the return data bus when addressed by the TFG (once per major frame).

I. SHUTTLE INTERFACE

1. The circuit design of this interface is dependent upon the ultimate requirements placed upon the payload by the shuttle

The SCCM is a mission peculiar module; however, a number of circuit types are required on every mission (solenoid and pyro firing circuits). As a result it is desirable to physically separate those circuits unique to EOS-A from those which are standard. This physical separation shall be accomplished using the following printed circuit boards (NOTE: more than one circuit board may be used for a given function, if required):

Solar Array Drive Control

Orbit Transfer Control

Structural Heater Control (twenty circuits)

Structural Thermistor Signal Conditioning (sixty circuits) and Module Lock Indication

Adapter Separation

Solar Array Deployment (less solenoid drivers)

Shuttle Interface Circuitry

Solenoid Drivers (2 firing busses; 10 firing circuits)

Pyro Drivers (not required on EOS-A)

3 Spare Boards (not required on EOS-A)

These twelve boards shall mate with a mother board unique to each mission.

3.2.2.1.2 Harnessing. All spacecraft harnessing shall be separated by function (power, command and telemetry data busses, timecode and clock frequencies, solenoid and pyro drive signals, stepper motor signals, heater and thermistor signals, digital sensor data signals, analog control signals, coax) for wrapping with copper tape shielding. Shields shall be tied to the chassis of the user subsystem for all signals less than 100 kHz, except for cables carrying currents in excess of 5 amps for periods less than 100 msec (pyro and solenoid drives) which shall have the external shield tied at both ends. Signals in excess of 100 kHz shall also have shields tied to chassis at both ends.

3.2.2.1.3 Command and Telemetry. Table 3-3 lists the number and type of commands required by each module in the EOS-A spacecraft. Table 3-4 lists the number and type of telemetry outputs of each module in the EOS-A spacecraft. These data shall be formatted into a 128 x 64 word major frame and data shall be transmitted to the ground at 4 kbps, giving a major frame period of 16.384 seconds.

3.2.2.1.4 Grounding. All components within each spacecraft module shall have their cases electrically tied to the module frame which is, in turn, electrically tied to the spacecraft frame. All components, with the exception

Table 3-3. EOS-A Commands

(TBD)

Table 3-4. EOS-A Telemetry

(TBD)

of RF devices, shall provide isolation between power, signal, and chassis grounds. Isolation between power and signal grounds shall also be provided by means of a DC/DC converter in the component power input circuit. All power grounds (primary return of DC/DC converter) shall be tied to a single power return point in the module. This power return point shall be tied to the spacecraft unipoint ground on the transition frame. All signal grounds within a module shall be tied to a single return point in the module which is tied directly to the spacecraft unipoint ground. Figure 3-2 shows the spacecraft grounding concept.

3.2.2.2 Mechanical

The SCCM shall be less than 600 cubic inches and shall weigh less than 20 pounds and be bolt mounted directly to the transition frame. The electrical harness supporting the basic spacecraft bus (external to the modules) shall weigh less than forty pounds. The harness supporting the mission peculiar equipment (wideband module, thematic mapper, MSS gimbal assemblies, TDRSS and STDN antennas) shall weigh less than twenty pounds.

3.2.2.3 Thermal

The SCCM shall be thermally isolated from the spacecraft structure. All harness runs on the spacecraft structure shall be covered with thermal blankets to protect them from the extreme environment experienced in orbital operation.

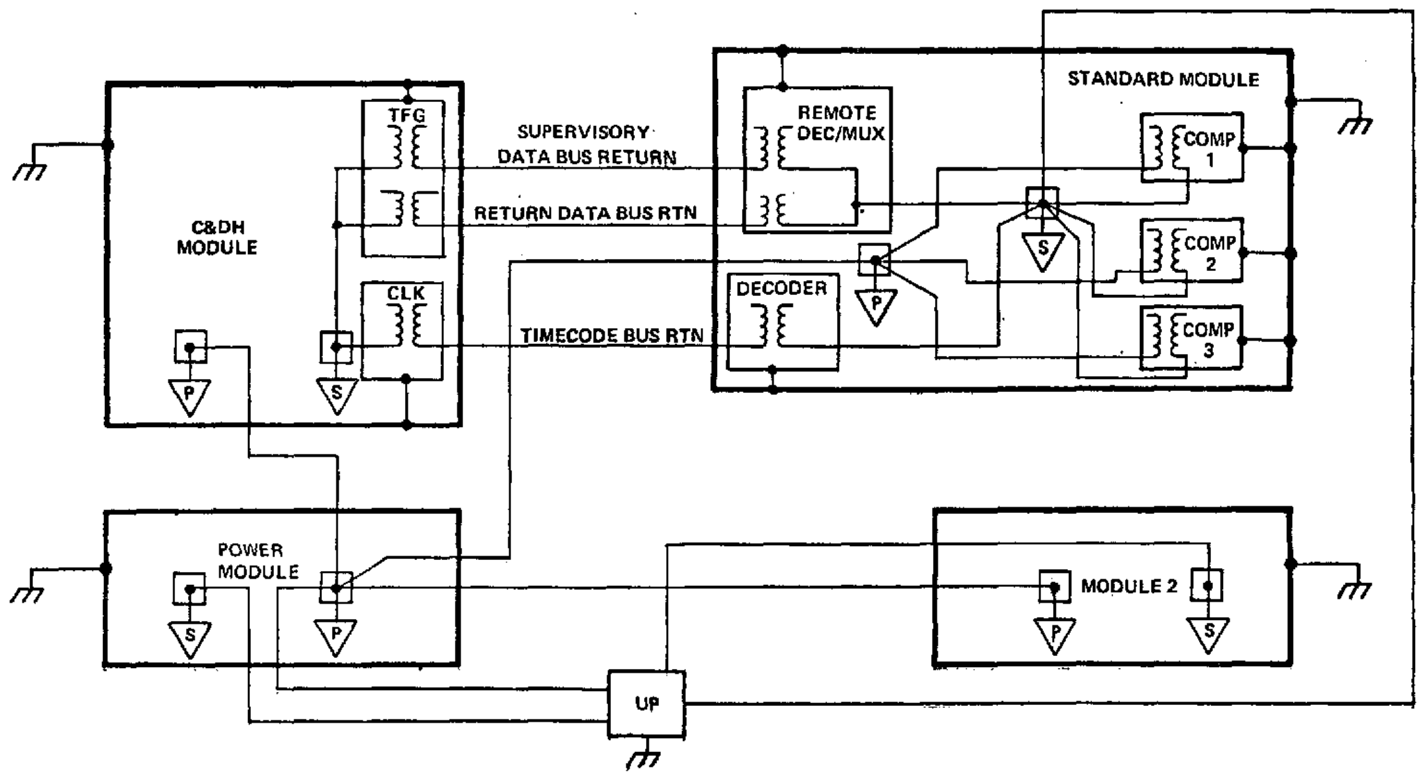

Figure 3-2. Grounding Concept